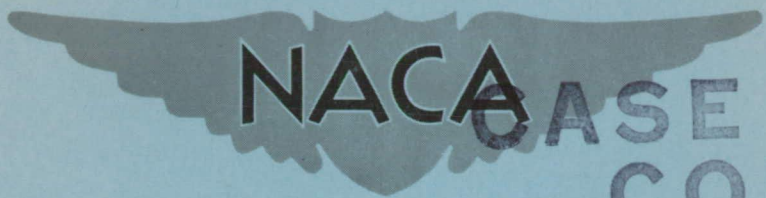


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# RESEARCH MEMORANDUM

TRANSONIC WIND-TUNNEL INVESTIGATION OF AERODYNAMIC-LOADING  
CHARACTERISTICS OF A 2-PERCENT-THICK TRAPEZOIDAL  
WING IN COMBINATION WITH BASIC  
AND INDENTED BODIES

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## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

January 23, 1957

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## AND INDENTED BODIES

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## SUMMARY

Results have been obtained in the Langley 8-foot transonic tunnel at Mach numbers from 0.80 to 1.115 and at angles of attack from  $0^\circ$  to  $20^\circ$  which indicate the static-aerodynamic-loads characteristics for wing-body configurations employing a 2-percent-thick trapezoidal wing in combination with basic and indented bodies.

These results show that the effects of indentation on the wing and body pressures are slight, as would be expected for this extremely thin wing configuration. Flow separation occurs over the outboard wing stations at relatively low angles of attack for subsonic Mach numbers and extends inboard with an increase in angle of attack. The spanwise distributions of load vary from approximately elliptical in shape at low angles of attack to more nearly triangular at the higher angles. Although lateral center-of-pressure movements resulting from changes in either angle of attack or Mach number are generally slight and gradual, the rearward movement of a main compression shock over the wing causes an abrupt rearward shift to occur in the chordwise center-of-pressure position as Mach number is increased from 0.90 to 0.98.

## INTRODUCTION

A general research program has recently been conducted in the Langley 8-foot transonic tunnel to determine the static-aerodynamic-loads characteristics for several wing-body combinations. Pressure distributions have been obtained at Mach numbers from 0.80 to about 1.12 and at angles of attack from  $0^\circ$  to  $20^\circ$  for trapezoidal, swept, and delta-wing combinations with and without body indentation.

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Results for the trapezoidal wing-body configurations are presented in this paper. The wing had  $26.6^\circ$  sweepback of the quarter-chord line, an aspect ratio of 2.61, a taper ratio of 0.211, and 2-percent-thick symmetrical circular-arc airfoil sections parallel to the plane of symmetry with the maximum thickness at the midchord point. Reynolds numbers, based on the mean aerodynamic chord, were on the order of  $2.6 \times 10^6$ .

#### SYMBOLS AND COEFFICIENTS

b	wing span
$b'/2$	unsupported semispan, distance from outer face of wing mounting block to wing tip
c	airfoil section chord, measured parallel to plane of symmetry
$c_{av}$	average wing chord, $S/b$
$\bar{c}$	wing mean aerodynamic chord
M	free-stream Mach number
$p_l$	local static pressure
p	free-stream static pressure
q	free-stream dynamic pressure
$r_{av}$	average body radius
S	wing area, total
x	distance measured streamwise from section leading edge
y	distance measured laterally from plane of symmetry
$y'$	distance measured laterally from outer face of wing mounting block
$\alpha$	angle of attack of body center line
$\Delta\alpha$	angle of attack of wing station minus angle of attack of body center line

$\frac{\partial \Delta \alpha}{\partial n}$	change in wing station twist angle due to normal force at wing section quarter chord
$\frac{\partial \Delta \alpha}{\partial m}$	change in wing station twist angle due to pitching moment about wing section quarter chord
$C_p$	pressure coefficient, $\frac{p_l - p}{q}$
$C_{p, \text{sonic}}$	pressure coefficient corresponding to local Mach number of unity
$c_n$	wing section normal-force coefficient, $\int_0^1 (C_{p_L} - C_{p_U}) d\left(\frac{x}{c}\right)$
$c_{m, c/4}$	wing section pitching-moment coefficient, taken about section quarter chord, $\int_0^1 (C_{p_L} - C_{p_U}) \left(0.25 - \frac{x}{c}\right) d\left(\frac{x}{c}\right)$
$C_N$	combination normal-force coefficient, based on total wing area, $C_{N,w} + C_{N,fw}$
$C_{m, \bar{c}/4}$	combination pitching-moment coefficient, based on total wing area and referred to $\bar{c}/4$ , $C_{m,w} + C_{m,fw}$
$C_{N,w}$	wing normal-force coefficient, based on total wing area
$C_{m,w}$	wing pitching-moment coefficient, based on total wing area and referred to $\bar{c}/4$
$C_{N,fw}$	normal-force coefficient for fuselage in presence of wing, based on total wing area
$C_{m,fw}$	pitching-moment coefficient for fuselage in presence of wing, based on total wing area and referred to $\bar{c}/4$
$C_b$	wing bending-moment coefficient, referred to body center line, $\int_{r_{av}}^{1.0} \left(\frac{c_{nc}}{c_{av}}\right) \left(\frac{y}{b/2}\right) d\left(\frac{y}{b/2}\right)$
$C_{N,f}$	normal-force coefficient for fuselage alone, based on total wing area

$C_{m,f}$  pitching-moment coefficient for fuselage alone, based on total wing area and referred to  $\bar{c}/4$

Subscripts:

L lower surface

U upper surface

## APPARATUS

### Tunnel

The Langley 8-foot transonic tunnel is a single-return, dodecagonal, slotted-throat wind tunnel designed to obtain aerodynamic data through the speed of sound while minimizing the usual effects of blockage. The tunnel, described more extensively in reference 1, operates at a stagnation pressure which is close to atmospheric.

### Models

Three-view drawings of the configurations tested are shown in figure 1. The wing of the combination was trapezoidal in plan form and had  $26.6^\circ$  sweepback of the quarter-chord line, an aspect ratio of 2.61, a taper ratio of 0.211, and 2-percent-thick symmetrical circular-arc airfoil sections parallel to the plane of symmetry with the maximum thickness located at the midchord station. The wing was constructed of type 416 stainless steel.

Two body configurations were tested in combination with the wing. The first, designated the basic body, was designed using Sears-Haack ordinates in order to obtain a low-drag body configuration at transonic speeds. The second or elliptical body retained the upper and lower basic body lines and was indented on the sides in the vicinity of the wing-body juncture to provide a desirable area distribution for a Mach number of 1.2. (See ref. 2.) Cross sections in the region of the indentation were made elliptical. (See fig. 1.) Design ordinates for the bodies are given in table I.

## MEASUREMENTS AND ACCURACY

Measurements of the local static pressures over the configurations were made by use of 137 orifices distributed over the upper and lower

wing surfaces at three wing semispan locations and along five fuselage meridian rows. Orifice locations are given in tables in figure 1. Pressure coefficients, determined from these measurements, are estimated to be accurate within  $\pm 0.005$  and are presented in tables II to IV.

Model angle of attack was measured by means of a fixed-pendulum strain-gage unit located in the nose of the model and is estimated to be accurate within  $\pm 0.1^\circ$ . Calibrations with tunnel empty indicate that local deviations from the average free-stream Mach number did not exceed 0.003 at subsonic speeds and did not become greater than about 0.01 as Mach number was increased to about 1.115. (See ref. 1.)

The wing was calibrated for deflection due to load by using an arrangement of a hydraulic jack and a balance scale to apply normal loads and an optical device to read vertical deflections of the wing leading and trailing edges. Wing twist angles, computed by using the experimental wing section data (table V) in conjunction with influence coefficients obtained from the measurements of wing deflection (table VI) are estimated to be accurate to about  $0.25^\circ$ . It should be noted that the wing section data (table V) and also the data for the basic body alone (table VII) have been copied directly from the results of the mechanical integrations to a number of decimal places generally used in the computational procedure. Probable accuracies of the normal-force and pitching-moment coefficients presented in tables V and VII are on the order of  $\pm 0.01$  and  $\pm 0.005$ , respectively.

## TESTS

The wing-body configurations and the basic body alone were tested at Mach numbers from 0.80 to 1.115 and through an angle-of-attack range extending generally from  $0^\circ$  to  $20^\circ$ .

For the present tests, transition strips were fixed on the model at 10 percent of the wing chord and at 10 percent of the body length. These strips, approximately 1/10 inch wide and composed of no. 120 carborondum grains set in a plastic adhesive at a medium density (about 30 grains per inch), extended from the wing-body juncture to the wing tip on the upper and lower wing surfaces and also formed a ring around the fuselage at 10 percent of the body length. Reynolds numbers for the tests were on the order of  $2.6 \times 10^6$ , based on the wing mean aerodynamic chord (fig. 2).

## CORRECTIONS

Effects of subsonic boundary interference in the slotted test section are considered negligible and no corrections for these effects have been applied. At some supersonic speeds, model pressure distributions would be affected by boundary-reflected compression and expansion waves; therefore, no data are presented for the Mach number range ( $M > 1.03$  to  $M < 1.115$ ) over which these effects might appear.

No corrections for the effects of wing aeroelasticity have been applied. In order to provide an indication of these effects, however, figure 3 has been prepared to show the spanwise twist distribution for the elliptical configuration at a Mach number of 1.115 and an angle of attack of  $20^\circ$ . As noted previously, the twist has been estimated by using influence coefficients obtained from the measurements of wing deflection in conjunction with the experimental wing section data.

The change in angle at a given wing semispan location may be determined from the expression:

$$\Delta\alpha = q \frac{b'}{2} \int_0^1 \left( \frac{\partial \Delta\alpha}{\partial n} c_{c_n} + \frac{\partial \Delta\alpha}{\partial m} c_{c_m}^2 \right) d \left( \frac{y'}{b'/2} \right)$$

Details concerning the loading procedure and the method used to obtain the influence coefficients may be found in reference 3.

## RESULTS

The following is an index of figures showing the model aerodynamic loading characteristics:

	Figure
Pressure coefficients for wing in presence of bodies . . . . .	4
Pressure coefficients for bodies in presence of wing . . . . .	5
Pressure coefficients for basic body alone . . . . .	6
Spanwise load distributions for two configurations . . . . .	7
Normal-force and pitching-moment characteristics of wing-body configurations . . . . .	8
Normal-force and pitching-moment characteristics of wing in presence of body . . . . .	9
Variation of wing bending-moment coefficient with normal- force coefficient for several Mach numbers . . . . .	10

## Figure

Lateral center-of-pressure position . . . . .	11
Chordwise center-of-pressure position . . . . .	12
Part of total load carried by wing . . . . .	13

Data points which were obtained for the elliptical-body configuration at an angle of attack of  $16^\circ$  for Mach numbers of 0.98, 1.03, and 1.115 are presented in figures 8 and 9 only. It should be noted that staggered scales have been used in many figures and care should be taken in selecting the proper zero axis for each curve.

## DISCUSSION

## Wing Pressure Distributions

At a Mach number of 0.80, a leading-edge separation vortex apparently has a predominant effect on the flow over the upper surface of the wing. The shapes of the pressure distributions at an angle of attack of  $4^\circ$  (fig. 4(a)) are indicative of such a vortex in that the upper-surface pressure peaks near the wing leading edge generally become lower and broader with increasing distance from the wing-body juncture. The pressure peak at the 20-percent semispan station is lower than that at the 40-percent semispan station probably because of the proximity of the 20-percent station to the body surface. (See fig. 1.) At an angle of attack between  $4^\circ$  and  $8^\circ$  (fig. 4(a)), the leading-edge separation vortex causes flow separation over the outboard part of the wing. Further increases in angle of attack are accompanied by an inboard spread of the separated region, with separation over the entire wing upper surface evident at an angle of  $20^\circ$ .

As Mach number is increased from 0.80, the effects of shock waves on the upper-surface flow become predominant. (See figs. 4(b) to 4(g).) These shock waves move rearward with increasing Mach number resulting in increased loading over the trailing-edge regions of the wing and a corresponding rearward shift of the wing center of pressure. The shocks are also conducive to flow separation initially occurring behind the shocks on the upper-surface outboard regions. However, since the shocks move rearward with increasing Mach number, the initial occurrence of flow separation is delayed to higher angles of attack as Mach number is increased. (See fig. 4.) Because this wing has a fair amount of leading-edge sweep, it is believed that the flow phenomena may be similar in many respects to those described in some detail for a swept wing at high subsonic and transonic speeds in reference 4.



Figure 4 shows that the effects of body indentation on the wing pressures are generally slight. Some large effects occur, however, at the two outboard stations for a Mach number of 0.90 at an angle of attack of  $12^\circ$  and for a Mach number of 0.94 at angles of attack of  $8^\circ$  and  $12^\circ$ . Reasons for these differences are not known. It is possible, however, that body indentation has a sufficient effect on the shocks to influence their position and therefore the shock-induced flow separation.

#### Body Pressure Distributions

Examination of the pressure distributions for the bodies in the presence of the wing (figs. 5(a) to 5(g)) indicates that the effects of indentation on the body pressures are slight at all Mach numbers and are generally confined to regions of local accelerations and decelerations of the flow corresponding to changes in the curvature of the indentation.

Increases in free-stream Mach number result in a pronounced rearward shift of the minimum pressure peak located on the upper part of the fuselage (see rows A and B, figs. 5(a) to 5(g)) while the location of the positive pressure peak on the lower part of the fuselage remains relatively unchanged (see rows D and E, figs. 5(a) to 5(g)). Because the flow over the body is so greatly influenced by the flow over the wing, it would be expected that the rearward movement of the main wing compression shock would have a considerable effect on the upper-surface body pressures. Similarly, the large positive pressures on the lower wing surface near the leading edge determine the position of the positive pressure peak on the lower part of the fuselage.

Comparison of the pressure distributions for the bodies in the presence of the wing (fig. 5) with those for the basic body alone (fig. 6) gives an indication of the increase in loading over the body resulting from wing interference. As a typical example, at a Mach number of 0.98 and an angle of attack of  $8^\circ$ , the basic body in the presence of the wing carries about nine times the load carried by the basic body alone. Force results for the basic body alone may be found in reference 5.

#### Spanwise Load Distributions

Spanwise load distributions for the two configurations (fig. 7) vary in shape from approximately elliptical at low angles of attack to more nearly triangular at the higher angles. Because only three semi-span points were used to establish the spanwise loading curves, it is possible that the faired curves presented in figure 7 may not represent the true spanwise distribution of load. Comparisons to be presented later between the results of these tests, however, and force results

also obtained in the Langley 8-foot transonic tunnel for identical configurations (at angles of attack up to  $6^\circ$ ) indicate close agreement of the normal-force coefficients.

Included in figure 7 on the span-load plots are the spanwise variations in loading over the fuselage for several Mach numbers. For a given angle of attack, the shape of the spanwise fuselage loading varies only slightly with a change in Mach number. At a constant Mach number, however, an increase in angle of attack was accompanied by a considerable increase in loading over the center part of the body.

As would be expected, only slight variations in the span loadings generally resulted from the change in body shape. The differences seen at  $M = 0.94$  for  $\alpha = 12^\circ$  are a result of the variations noted in the wing pressure distributions of figure 4(c).

Wing-body juncture locations shown in figure 7 for the basic and elliptical configurations were obtained by taking a root-mean-square value of the fuselage width over the region of the fuselage intersected by the wing. The resulting values were  $0.175b/2$  and  $0.165b/2$  for the basic and elliptical bodies, respectively.

#### Normal-Force and Pitching-Moment Characteristics

Wing-body configuration.- With the exception of the higher angles of attack at Mach numbers of 0.90 and 0.94, the effects of body indentation on the normal-force and pitching-moment characteristics are slight. (See fig. 8.) The differences appearing at Mach numbers of 0.90 and 0.94 are again due to the disturbances appearing in the wing pressure distributions. (See figs. 4(b) and 4(c).)

Comparison of the results of these tests (fig. 8) with results of force tests noted previously of an identical configuration (obtained at angles of attack up to  $6^\circ$ ) indicates good agreement in the normal-force results. Similarly, good agreement is also shown in the comparison of pitch characteristics.

Wing.- Comparison of the results for the wing in the presence of the bodies (fig. 9) with those for the wing-body combinations (fig. 8) shows an expected decrease in slope of the normal-force curve and an increase in stability at low angles of attack for the wing with interference. It is also of interest to note that the destabilizing breaks, seen at high normal-force coefficients and Mach numbers for the wing-body configurations (fig. 8), do not appear in the curves for the wing in the presence of the bodies (fig. 9). Examination of the pressure distributions for the bodies in the presence of the wing (figs. 5(e) to 5(g)) indicates that the destabilizing break is probably associated with

a forward shift in the center of pressure for the bodies which occurs at the highest angle of attack.

### Wing Bending Moments

Wing bending-moment coefficients (referred to the body center line) indicate only slight variations due to body indentation (fig. 10). The highest bending-moment coefficient obtained in these tests occurred for the elliptical configuration and progressed from a value of 0.30 at a Mach number of 0.80 to a peak value of 0.51 at  $M = 0.98$  and decreased thereafter to a value of 0.475 at  $M = 1.115$ .

### Center-of-Pressure Location

Lateral position.- As would be expected from the results presented in figures 9 and 10, body indentation has only a slight ( $\pm 1$  percent) effect on the lateral position of the center of pressure (fig. 11). It should be noted that lateral center-of-pressure positions presented in this paper have been determined for the exposed wing and are referred to the body center line.

Variations in the lateral center-of-pressure location with an increase in wing normal-force coefficient are gradual and amount to an inboard shift of about 5 percent of the semispan at a Mach number of 0.80 and a 1-percent inboard shift at  $M = 1.115$ . Estimates of the lateral center of pressure have been made by using the theories of references 6 and 7. Although the theories noted do not include the effects of wing-body interference, it may be seen (fig. 11(a)) that agreement in theory and experiment is excellent, the lateral center-of-pressure position being predicted within about 1 percent of the semispan at low normal-force coefficients. Variations in the lateral center-of-pressure position with a change in Mach number (fig. 11(b)) indicate random movements on the order of  $\pm 1$  percent of the semispan at wing normal-force coefficients at 0.25 and 0.40. At a wing normal-force coefficient of 0.60, a 5-percent outboard shift occurs between Mach numbers of 0.80 and 0.94, and thereafter the center-of-pressure location remains fairly constant.

Chordwise position.- Variation of the chordwise center-of-pressure position (for the exposed wing) with an increase in wing normal-force coefficient is illustrated in figure 12(a). At the lower Mach numbers, it may be seen that the center of pressure moves steadily rearward with an increase in normal force, followed by an abrupt further rearward movement at the higher wing normal-force coefficients. The abrupt rearward movement is a result of the increase in the extent of separation

occurring at angles of attack of  $12^\circ$  and  $20^\circ$  for the wing. (See figs. 4(a) to 4(c).) Overall movement of the center-of-pressure position at these subsonic Mach numbers is on the order of 15 percent of the wing mean aerodynamic chord. At the higher Mach numbers ( $M = 0.98$  to  $M = 1.115$ ) the rearward movement becomes more gradual and the overall movement is considerably reduced.

Variations in the chordwise center-of-pressure position with Mach number (fig. 12(b)) indicate an abrupt rearward movement occurring between Mach numbers of 0.90 and 0.98. Examination of the wing pressure distributions for these Mach numbers shows that the abrupt rearward movement is caused by the rearward movement of the main compression shock over the wing and the accompanying increase in loading over the trailing-edge regions of the wing. As would be expected, body indentation generally had only a slight effect on the chordwise center-of-pressure position.

#### Division of Load

Figure 13 indicates the part of the total load carried by the wing of each configuration. At low normal-force coefficients, the wing of the basic configuration carries about 78 percent of the total load, whereas that of the elliptical configuration carries approximately 80 percent. The slight increase in load carried by the wing of the elliptical configuration is due primarily to the slight increase in exposed wing area for this configuration. Onset of wing stall at a Mach number of 0.80 results in a fairly abrupt dropoff in the percent load carried by the wing at high normal-force coefficients. At the higher Mach numbers, the wing approaches the stalled condition more gradually (see fig. 9) and the dropoff in the load proportion carried by the wing shown in figure 13 becomes more gradual. Estimates of the load division were made by using the method described in reference 8, and it may be seen that agreement in the calculated and experimental results is good.

#### CONCLUSIONS

The results of tests of a thin trapezoidal wing in combination with both basic and indented bodies have led to the following conclusions:

1. The effects of indentation on both wing and body pressures are slight.
2. For subsonic Mach numbers wing separation first occurs over the outboard wing stations at relatively low angles of attack and extends inboard with an increase in angle of attack.

3. Spanwise load distributions vary from approximately elliptical in shape at low angles of attack to more nearly triangular at the higher angles.

4. Although lateral center-of-pressure movements resulting from changes in either angle of attack or Mach number are both slight and gradual, the rearward movement of the main compression shock over the wing causes an abrupt rearward shift to occur in the chordwise center-of-pressure position as Mach number is increased from 0.90 to 0.98.

Langley Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., September 26, 1956.

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TABLE I

## BODY ORDINATES

Body station, inches from nose	Basic body radius, in.	Elliptical body	
		Semimajor axis, in.	Semiminor axis, in.
0	0	Circular; radius equal to basic body radius  ↓	
1.0	.282		
2.0	.460		
3.0	.612		
4.0	.743		
6.0	.969		
8.0	1.150		
10.0	1.290		
12.0	1.404		
13.426	1.475		
14.0	1.493	1.493	1.475
16.0	1.552	1.552	1.473
18.0	1.590	1.590	1.437
20.0	1.606	1.606	1.434
22.0	1.594	1.594	1.463
24.0	1.560	1.560	1.524
26.0	1.501	Circular; radius equal to basic body radius  ↓	
28.0	1.414		
30.0	1.300		
32.0	1.158		
34.0	.984		
36.15	.750		

Basic body maximum radius = 1.606 inches at body station 20.10 inches.

TABLE II  
PRESSURE COEFFICIENTS FOR WING IN THE PRESENCE OF THE BODIES

M = 0.80

x c	Basic						Elliptical						x c
	Upper surface			Lower surface			Upper surface			Lower surface			
	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	
$\alpha = 0^\circ$													
.050	-.045	-.036		-.021	.004		-.014	-.099		.001	.044		.050
.075	-.036			-.029	.003		-.017			-.010	.025		.075
.100		-.026			.002			-.023	.042		.009		.100
.150	-.063	-.052	-.049	-.064	-.056	-.115	-.019	-.051	-.036	-.027		.023	.150
.250	-.074	-.082	-.067	-.083	-.078		-.022	-.058	-.066	-.044	-.050		.250
.350	-.083	-.125	-.094	-.083	-.078	-.096	-.027	-.079	-.104	-.055	-.065	-.075	.350
.450	-.108	-.115	-.115	-.111	-.097	-.098	-.080			-.063	-.072	-.081	.450
.550	-.112	-.128	-.118	-.133	-.114	-.100	-.085	-.097	-.095	-.062	-.062	-.087	.550
.650	-.105	-.111	-.108	-.123	-.100	-.090	-.067	-.089	-.089	-.074	-.055	-.061	.650
.750	-.101		-.124	-.094	-.089	-.082	-.067		-.064			-.050	.750
.825			-.078			-.069				-.030	-.036		.825
.850	-.080	-.061		-.073	-.063		-.041			-.027			.850
.900	-.056	-.042		-.058			-.037	-.033				-.014	.900
.950	-.029	-.015				-.024	-.033	-.014					.950
$\alpha = 4^\circ$													
.050	-.276	-.677		.206	.235		-.307	-.702		.189	.249		.050
.075	-.245			.175	.207		-.283			.150	.214		.075
.100		-.656	-.724	.171	.201			-.640	-.746		.214		.100
.150	-.214	-.529	-.662	.171	.201		-.229	-.482	-.588		.172	.221	.150
.250	-.213	-.234	-.463	.087	.099		-.162	-.217	-.367	.112	.111		.250
.350	-.194	-.213	-.240	.045	.052	.047	-.138	-.182	-.250	.064	.066	.051	.350
.450	-.193	-.172	-.145	.001	.008	.013	-.184			.039	.029	.016	.450
.550	-.164	-.162	-.135	-.029	-.019	-.013	-.155	-.174	-.154	.014	.004	-.013	.550
.650	-.137	-.130	-.115	-.039	-.023	-.024	-.117	-.134	-.126	-.004		-.036	.650
.750	-.117		-.095	-.033	-.033	-.030	-.099	-.114	-.138	-.012	-.027	-.027	.750
.825			-.077			-.027			-.087			-.026	.825
.850	-.081	-.058		-.030	-.024		-.059			-.003	-.017		.850
.900	-.053	-.039		-.017		-.003	-.045	-.044		-.010		-.005	.900
.950	-.025	-.007					-.036	-.027					.950
$\alpha = 8^\circ$													
.050	-.822	-1.211		.392	.406		-1.003	-1.189		.393	.416		.050
.075	-.743			.346	.357		-.809			.351	.368		.075
.100	-.687	-1.252	-.653	.307	.307	.336		-1.208	-.664		.320		.100
.150	-.435	-1.232	-.642	.307	.307	.336	-.473	-1.171	-.652		.320	.353	.150
.250	-.353	-.939	-.614	.218	.215	.211	-.311	-.942	-.630	.233	.233		.250
.350	-.307	-.240	-.578	.153	.150	.141	-.253	-.265	-.591	.169	.167	.150	.350
.450		-.224		.086	.097	.083	-.271			.125	.113	.095	.450
.550	-.263	-.248	-.515	.047	.056		-.237	-.213	-.502	.085	.070	.047	.550
.650	-.227	-.214	-.483	.016	.034	-.015	-.180	-.189	-.462	.050		-.004	.650
.750	-.197	-.177	-.442	-.001	-.012	-.039	-.150	-.160	-.406	.022	.005	-.028	.750
.825			-.416			-.058			-.389			-.048	.825
.850	-.130			-.016	-.020		-.096			.016	-.003		.850
.900	-.095			-.013		-.100	-.077	-.081		.005		-.089	.900
.950	-.052	-.054					-.054	-.047					.950
$\alpha = 12^\circ$													
.050	-1.467	-1.012		.547	.530		-1.433	-1.042		.536	.530		.050
.075	-1.324			.494	.476		-1.341			.484	.477		.075
.100	-1.159	-.968	-.652	.422	.422	.461	-1.250	-.988	-.668	.427	.427		.100
.150	-.804	-.922	-.650	.331	.321	.309	-.978	-.953	-.684	.333	.325	.463	.150
.250	-.560	-.876	-.639	.251	.242	.225	-.549	-.879	-.654	.255	.247	.310	.250
.350	-.461	-.825	-.626	.170	.176	.158	-.407	-.815	-.641	.190	.180	.224	.350
.450		-.737		.115	.119		-.390			.141	.121	.155	.450
.550	-.377	-.645	-.598	.066	.046	.024	-.353	-.636	-.606	.089	.038	.093	.550
.650	-.345	-.552	-.576	.021	.006	-.023		-.544	-.583			.022	.650
.750	-.309	-.468	-.558			-.055	-.276	-.467	-.587	.017	-.021	.750	
.825			-.541						-.547		-.056	.825	
.850	-.225			-.026	-.043		-.202			.004	-.034		.850
.900	-.182			-.041		-.135	-.174	-.343		-.026		-.134	.900
.950	-.130	-.287					-.145	-.289					.950
$\alpha = 20^\circ$													
.050	-.555	-.515		.757	.688		-.548	-.525		.762	.697		.050
.075	-.552			.703	.637		-.547			.700	.644		.075
.100	-.552	-.521	-.556	.582	.582	.551	-.548	-.530	-.549		.598		.100
.150	-.552	-.525	-.557	.502	.481	.444	-.551	-.533	-.551	.513	.489	.580	.150
.250	-.561	-.533	-.559	.403	.391	.353	-.564	-.542	-.556	.415	.403	.361	.250
.350	-.572	-.545		.311	.308	.275	-.579	-.554	-.562	.333	.319	.281	.350
.450	-.590	-.560	-.593	.236	.236	.195	-.597			.262	.245	.203	.450
.550	-.606	-.575	-.607	.149	.160	.110	-.613	-.585	-.592	.185	.177	.119	.550
.650	-.616	-.589	-.621	.068	.069	.039	-.599	-.599	-.609	.092	.083	.050	.650
.750	-.627	-.603	-.628			-.006	-.631	-.607	-.629			.001	.750
.825			-.628							.002	-.013		.825
.850	-.633	-.604		-.045	-.027		-.631			-.074	-.077	-.108	.850
.900	-.628	-.600		-.104	-.091	-.115	-.632	-.605					.900
.950	-.622	-.591					-.626	-.596					.950



TABLE II.- Continued  
PRESSURE COEFFICIENTS FOR WING IN THE PRESENCE OF THE BODIES

M = 0.90

Basic						
h/b	Upper surface			Lower surface		
	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2
$\alpha = 0^\circ$						
.050	.001	-.013		.011	.026	
.075	.004			.016	.025	
.100		-.002			.010	-.108
.150	-.021	-.041	-.057			
.250	-.061	-.075	-.066	-.048	-.046	
.350	-.075	-.122	-.104	-.065	-.078	-.106
.450	-.111	-.118	-.139	-.106	-.105	-.116
.550	-.118	-.138	-.137	-.143	-.129	-.120
.650	-.114	-.123	-.120	-.133	-.112	-.110
.750	-.104		-.104	-.092	-.095	-.087
.825			-.084			-.068
.850	-.073	-.051		-.058	-.055	
.900	-.044	-.030		-.034		-.017
.950	-.016	.000				
$\alpha = 4^\circ$						
.050	-.294	-.684		.208	.241	
.075	-.256			.181	.215	
.100		-.451	-.838		.181	
.150	-.219	-.354	-.593		.105	.201
.250	-.257	-.309	-.353	.096	.105	
.350	-.262	-.321	-.296	.054	.054	.042
.450	-.273	-.279	-.221	.003	.009	-.001
.550	-.225	-.219	-.156	-.043	-.035	-.038
.650	-.157	-.146	-.126	-.060	-.044	-.053
.750	-.120		-.101	-.051	-.054	-.056
.825			-.085			-.048
.850	-.074	-.053		-.041	-.037	
.900	-.044	-.032		-.024		-.012
.950	-.017	.000				
$\alpha = 8^\circ$						
.050	-.752	-1.186		.408	.417	
.075	-.573			.363	.369	
.100	-.565	-1.111	-1.020		.321	
.150	-.507	-1.030	-.987		.235	.349
.250	-.459	-.889	-.887	.239	.235	.228
.350	-.452	-.805	-.805	.173	.168	.154
.450		-.420		.102	.111	.094
.550	-.455	-.430	-.613	.056	.062	
.650	-.374	-.387	-.523	.018	.022	-.019
.750	-.219	-.186	-.440	-.010	-.022	-.042
.825			-.391		-.056	
.850	-.107			-.025	-.038	
.900	-.071			-.019		-.081
.950	-.031	-.062				
$\alpha = 12^\circ$						
.050	-1.345	-1.020		.562	.540	
.075	-.901			.509	.487	
.100	-.805	-1.017	-.852		.436	.473
.150	-.681	-.962	-.861		.338	.323
.250	-.568	-.902	-.810	.351	.263	.244
.350	-.542	-.857	-.764	.270	.196	.174
.450		-.802		.187	.137	
.550	-.530	-.717	-.722	.132	.137	
.650	-.519	-.628	-.688	.077	.050	.041
.750	-.481	-.556	-.644	.022	.016	-.003
.825			-.627			-.029
.850	-.309			-.039	-.038	
.900	-.242			-.070		-.081
.950	-.185	-.389				
$\alpha = 20^\circ$						
.050	-.654	-.547		.798	.730	
.075	-.649			.749	.681	
.100	-.650	-.552	-.642		.630	
.150	-.647	-.555	-.644		.592	
.250	-.655	-.562	-.645	.551	.530	.491
.350	-.661	-.573		.455	.446	.409
.450	-.665	-.584		.369	.367	.335
.550	-.659	-.597	-.674	.299	.302	.263
.650	-.652	-.609	-.686	.218	.232	.183
.750	-.662	-.618	-.692	.146	.151	.122
.825			-.702			.086
.850	-.670	-.622		.051	.071	
.900	-.660	-.622		.005	.019	.003
.950	-.641	-.619				

Elliptical						
Upper surface			Lower surface			x/c
0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	
$\alpha = 0^\circ$						
-.008	-.117		.009	.057		.050
-.008			-.001	.036		.075
	-.014	.047		.006		.100
-.012	-.050	-.042	-.001	-.001	.026	.150
-.015	-.059	-.073	-.017	-.023		.250
-.022	-.085	-.123	-.044	-.053	-.089	.350
-.093			-.060	-.077	-.100	.450
-.104	-.126	-.123	-.074	-.090	-.107	.550
-.082	-.113	-.114	-.077		-.113	.650
-.080	-.093	-.088	-.064	-.076	-.077	.750
		-.079			-.060	.825
-.044			-.030	-.039		.850
-.038	-.033		-.025		-.013	.900
-.031	-.011					.950
$\alpha = 4^\circ$						
-.347	-.694		.202	.259		.050
-.328			.175	.224		.075
	-.310	-.840		.182		.100
-.232	-.291	-.504		.123	.232	.150
-.188	-.273	-.317	.123	.123		.250
-.172	-.264	-.302	.073	.072	.050	.350
-.244			.042	.030	.006	.450
-.214	-.228	-.165	.011	-.005	-.030	.550
-.138	-.158	-.136	-.016		-.059	.650
-.105	-.119	-.126	-.023	-.039	-.047	.750
		-.089			-.041	.825
-.053			-.008	-.021		.850
-.042	-.037		-.012		-.009	.900
-.031	-.011					.950
$\alpha = 8^\circ$						
-.847	-1.175		.399	.421		.050
-.664			.360	.374		.075
	-1.094	-1.039		.331	.363	.100
-.550	-.995	-.929		.242		.150
-.452	-.775	-.894	.248	.242		.250
-.378	-.574	-.811	.180	.180	.160	.350
-.432			.137	.121	.099	.450
-.463	-.449	-.619	.091	.075	.045	.550
-.314	-.366	-.527	.048		-.012	.650
-.159	-.193	-.425	.015	-.004	-.034	.750
		-.394			-.052	.825
-.090			.008	-.018		.850
-.070	-.087		-.005		-.080	.900
-.050	-.052					.950
$\alpha = 12^\circ$						
-1.316	-1.111		.551	.542		.050
-1.046			.500	.492		.075
-.833	-1.059	-.946		.444		.100
-.734	-.996	-1.025		.347	.476	.150
-.559	-.900	-.923	.357	.347	.337	.250
-.480	-.825	-.850	.277	.269	.244	.350
-.484			.215	.202	.179	.450
-.512	-.728	-.746	.162	.145	.111	.550
	-.643	-.702	.104		.043	.650
-.425	-.572	-.714	.046	.032	.002	.750
		-.637			-.027	.825
-.282			.001	-.021		.850
-.245	-.441		-.043		-.081	.900
-.211	-.390					.950
$\alpha = 20^\circ$						
-.654	-.565		.804	.742		.050
-.650			.745	.691		.075
-.650	-.570	-.643		.645		.100
-.651	-.572	-.644		.624		.150
-.663	-.579	-.648	.562	.543	.420	.250
-.675	-.588	-.651	.469	.458	.347	.350
-.677			.392	.380	.272	.450
-.671	-.610	-.674	.327	.314	.226	.550
-.623	-.684	.258	.266	.195	.150	.650
-.666	-.628	-.683	.175	.168	.135	.750
		-.697			.096	.825
-.666			.102	.087		.850
-.664	-.630		.037	.037	.012	.900
-.651	-.627					.950

TABLE II.- Continued  
PRESSURE COEFFICIENTS FOR WING IN THE PRESENCE OF THE BODIES

M = 0.94

Basic							Elliptical						
X c	Upper surface			Lower surface			Upper surface			Lower surface			X c
	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	
$\alpha = 0^\circ$													
.050	-.010	-.035		.021	.030		-.003	-.131		.015	.068		.050
.075	-.004			.033	.031		-.003			.006	.044		.075
.100		.001			.011	-.106		-.004	.054		.005	.029	.100
.150	-.031	-.040	-.081	-.077	-.048		-.005	-.046	-.041	-.012	-.021		.150
.250	-.064	-.081		-.077	-.048		-.008	-.056	-.076	-.067	-.088	-.123	.250
.350	-.078	-.139	-.120	-.064	-.086	-.127	-.017	-.087	-.135	-.044	-.055	-.101	.350
.450	-.127	-.141	-.177	-.116	-.123	-.148	-.102			-.067	-.088	-.123	.450
.550	-.154	-.184	-.205	-.175	-.168	-.177	-.136	-.158	-.172	-.092	-.116	-.143	.550
.650	-.168	-.201	-.198	-.192	-.170	-.184	-.113	-.165	-.158	-.109	-.116	-.160	.650
.750	-.154		-.191	-.137	-.150	-.146	-.102	-.111	-.108	-.082	-.098	-.098	.750
.825			-.121			-.101			-.090			-.068	.825
.850	-.091	-.071		-.075	-.074		-.046			-.032	-.043		.850
.900	-.055	-.038		-.051		-.021	-.036	-.028		-.022		-.010	.900
.950	-.019	-.005					-.029	-.005					.950
$\alpha = 4^\circ$													
.050	-.258	-.736		.212	.239		-.316	-.823		.209	.265		.050
.075	-.244			.188	.216		-.310			.187	.229		.075
.100		-.219	-.751		.182	.181		-.305	-.831		.229		.100
.150	-.204	-.273	-.599		.105		-.241	-.283	-.601		.187	.235	.150
.250	-.235	-.284	-.364	.099	.105		-.182	-.275	-.339	.129	.127		.250
.350	-.245	-.314	-.336	.057	.053	.033	-.156	-.256	-.322	.078	.077	.051	.350
.450	-.277	-.302	-.365	.001	.000	-.018	-.244			.004	.029	-.004	.450
.550	-.308	-.348	-.388	-.058	-.055	-.075	-.308	-.315	-.374	-.006	-.013	-.056	.550
.650	-.331	-.373	-.376	-.091	-.076	-.119	-.300	-.355	-.350	-.030		-.110	.650
.750	-.278		-.192	-.081	-.096	-.122	-.212	-.270	-.167	-.041	-.063	-.090	.750
.825			-.110			-.087			-.088			-.065	.825
.850	-.068	-.034		-.053	-.052		-.042			-.013	-.029		.850
.900	-.026	-.001		-.026		-.020	-.020	-.002		-.010		-.010	.900
.950	.003	.026					-.009	.018					.950
$\alpha = 8^\circ$													
.050	-.687	-1.071		.412	.417		-.749	-1.191		.410	.427		.050
.075	-.499			.369	.370		-.604			.372	.380		.075
.100	-.492	-.974	-1.190		.325	.355		-1.067	-1.111		.338	.363	.100
.150	-.445	-.896	-1.132		.227		-.499	-.742	-1.043		.253		.150
.250	-.405	-.740	-1.051	.243	.239		-.397	-.514	-1.003	.261	.253		.250
.350	-.411	-.572	-.919	.178	.172	.155	-.331	-.481	-.933	.190	.190	.164	.350
.450		-.486		.107	.114	.094	-.405			.144	.127	.102	.450
.550	-.471	-.509	-.641	.057	.066		-.481	-.488	-.636	.096	.079	.041	.550
.650	-.513	-.531	-.626	.012	.051	-.034	-.473	-.517	-.612	.046		-.024	.650
.750	-.570	-.561	-.606	-.028	-.036	-.069	-.494	-.531	-.530	.003	-.013	-.056	.750
.825			-.527			-.087			-.504			-.075	.825
.850	-.367	-.607		-.052	-.061		-.452			-.006	-.035		.850
.900	-.219	-.441		-.049		-.099	-.265	-.364		-.023		-.093	.900
.950	-.130						-.136	-.138					.950
$\alpha = 12^\circ$													
.050	-1.194	-1.281		.577	.555		-1.156	-1.285		.570	.557		.050
.075	-.752			.525	.505		-.894			.521	.506		.075
.100	-.727	-1.234	-1.067		.451	.485	-.811	-1.215	-1.369		.460	.486	.100
.150	-.656	-1.165	-1.059		.357	.341	-.701	-1.148	-1.308		.365	.352	.150
.250	-.595	-1.049	-1.013	.369	.357		-.543	-1.024	-1.233	.375	.365		.250
.350	-.538	-.971	-.975	.290	.285	.262	-.463	-.914	-1.206	.296	.292	.264	.350
.450		-.781		.207	.219	.196	-.502			.236	.228	.199	.450
.550	-.583	-.542	-.876	.159	.162		-.552	-.685	-.910	.185	.173	.136	.550
.650	-.611	-.520	-.820	.101	.075	.060		-.642	-.881	.128		.066	.650
.750	-.644	-.543	-.756	.047	.042	.017	-.616	-.661	-.832	.073	.061	.024	.750
.825			-.737			-.009			-.781			-.002	.825
.850	-.508			-.015	-.008		-.472			.033	.014		.850
.900	-.398	-.438		-.047		-.051	-.434	-.686		-.012		-.045	.900
.950	-.311						-.427	-.572					.950
$\alpha = 20^\circ$													
.050	-.817	-.801		.830	.762		-.819	-.813		.828	.763		.050
.075	-.812			.780	.712		-.817			.767	.716		.075
.100	-.811	-.805	-.808		.662	.625	-.816	-.821	-.809		.669	.644	.100
.150	-.811	-.809	-.807		.567	.527	-.813	-.819	-.800		.570		.150
.250	-.816	-.812	-.807	.585	.567		-.833	-.824	-.810	.589	.570		.250
.350	-.806	-.818		.493	.485	.448	-.823	-.824	-.806	.497	.490	.450	.350
.450	-.761	-.823		.405	.410	.378	-.783			.422	.413	.379	.450
.550	-.739	-.827	-.812	.343	.346	.308	-.753	-.817	-.813	.364	.351	.309	.550
.650	-.744	-.833	-.813	.267	.280	.231		-.828	-.817	.295	.311	.233	.650
.750	-.736	-.833	-.816	.197	.203	.175	-.736	-.822	-.787	.218	.212	.178	.750
.825			-.821			.139			-.816			.140	.825
.850	-.746	-.826		.110	.127		-.731			.152	.139		.850
.900	-.715	-.822		.068	.083	.064	-.722	-.818		.092	.093	.064	.900
.950	-.670	-.819					-.688	-.808					.950

TABLE II.- Continued  
PRESSURE COEFFICIENTS FOR WING IN THE PRESENCE OF THE BODIES

M = 0.98

Basic							Elliptical						
$\frac{x}{c}$	Upper surface			Lower surface			Upper surface			Lower surface			$\frac{x}{c}$
	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	
$\alpha = 0^\circ$													
.050	.032	.003		.035	.048		.017	-.101		.032	.093		.050
.075	.036			.045	.048		.019			.024	.066		.075
.100		.030			.009		.019	.030	.082	.024	.066		.100
.150	.008	-.013	-.056		.009	-.082	.019	-.021	-.015	.024	.066		.150
.250	-.037	-.052	-.039	-.027	-.024		.019	-.027	-.045	.012	.005	.055	.250
.350	-.049	-.101	-.090	-.037	-.061	-.101	.016	-.058	-.104	-.021	-.029	-.080	.350
.450	-.097	-.108	-.144	-.087	-.098	-.126	-.074			-.043	-.066	-.108	.450
.550	-.127	-.152	-.183	-.154	-.146	-.161	-.123	-.139	-.170	-.078	-.104	-.138	.550
.650	-.159	-.185	-.206	-.192	-.163	-.191	-.121	-.174	-.199	-.117		-.185	.650
.750	-.201		-.227	-.206	-.201	-.223	-.164	-.213	-.212	-.153	-.170	-.200	.750
.825			-.232			-.241			-.216			-.210	.825
.850	-.245	-.238		-.233	-.242		-.183			-.171	-.200		.850
.900	-.249	-.238		-.235			-.192	-.212		-.162		-.208	.900
.950	-.183	-.209				-.240	-.151	-.144					.950
$\alpha = 4^\circ$													
.050	-.210	-.708		.236	.259		-.269	-.739		.230	.288		.050
.075	-.205			.212	.236		-.266			.205	.254		.075
.100		-.154	-.662	.212	.236		-.209	-.245	-.721	.205	.254		.100
.150	-.171	-.225	-.582	.205	.205	.190	-.162	-.246	-.659	.212	.257		.150
.250	-.203	-.241	-.330	.123	.130		-.162	-.242	-.315	.156	.153		.250
.350	-.216	-.282	-.295	.083	.078	.062	-.128	-.228	-.287	.106	.103	.076	.350
.450	-.245	-.262	-.322	.028	.028	.013	-.209			.069	.055	.023	.450
.550	-.272	-.306	-.345	-.032	-.033	-.046	-.269	-.271	-.325	.027	.008	-.032	.550
.650	-.297	-.328	-.356	-.080	-.062	-.104	-.265	-.312	-.345	-.025		-.103	.650
.750	-.340		-.314	-.116	-.117	-.160	-.298	-.346	-.318	-.069	-.087	-.141	.750
.825			-.312			-.187			-.297			-.163	.825
.850	-.367	-.376		-.166	-.167		-.307			-.096	-.134		.850
.900	-.353	-.370		-.185		-.204	-.313	-.352		-.115		-.184	.900
.950	-.267	-.282					-.276	-.258					.950
$\alpha = 8^\circ$													
.050	-.623	-.1010		.438	.442		-.651	-.1094		.432	.450		.050
.075	-.432			.396	.396		-.533			.393	.405		.075
.100	-.427	-.835	-.1078	.396	.396		-.435	-.969	-.1018	.393	.405		.100
.150	-.383	-.741	-.1027	.274	.352	.381	-.339	-.668	-.971	.288	.363	.390	.150
.250	-.353	-.592	-.949	.209	.267	.257	-.339	-.452	-.923	.219	.217	.195	.250
.350	-.361	-.490	-.913	.209	.203	.187	-.274	-.421	-.880	.173	.159	.134	.350
.450		-.448		.137	.146	.127	-.345			.173	.159	.134	.450
.550	-.419	-.470	-.576	.088	.095		-.405	-.435	-.570	.123	.108	.073	.550
.650	-.460	-.491	-.558	.040	.078	-.008	-.443	-.465	-.548	.071	.071	.002	.650
.750	-.520	-.515	-.549	.012	-.015	-.048	-.445	-.490	-.513	.021	.003	-.036	.750
.825			-.513			-.071			-.485			-.058	.825
.850	-.515			-.062	-.062		-.459			-.002	-.033		.850
.900	-.532			-.079		-.095	-.460	-.511		-.035		-.083	.900
.950	-.439	-.518					-.466	-.493					.950
$\alpha = 12^\circ$													
.050	-.1087	-.1182		.241	.575		-.1050	-.1185		.599	.585		.050
.075	-.662			.390	.524		-.807			.552	.536		.075
.100	-.642	-.1119	-.1244	.476	.476	.507	-.723	-.1100	-.1235	.491	.491	.514	.100
.150	-.576	-.1048	-.1165	.384	.366		-.620	-.1036	-.1184	.398	.380	.250	.150
.250	-.485	-.942	-.1082	.318	.384	.366	-.476	-.916	-.1117	.330	.325	.300	.250
.350	-.473	-.882	-.959	.237	.313	.292	-.401	-.815	-.1092	.272	.265	.236	.350
.450		-.740		.218	.251	.230	-.438			.224	.211	.174	.450
.550	-.532	-.587	-.836	.195	.197		-.489	-.621	-.844	.168	.106	.106	.550
.650	-.577	-.566	-.813	.138	.116	.097	-.566	-.603	-.796	.113	.101	.066	.650
.750	-.641	-.607	-.741	.082	.079	.053			-.721			.041	.750
.825			-.684			.029	-.567			.079	.055		.825
.850	-.629	-.600		.027	.034		-.570	-.624		.035		.001	.850
.900	-.639			-.002		-.008	-.527	-.624					.900
.950	-.532												.950
$\alpha = 20^\circ$													
.050	1.234			.848	.780		-.1278	-.1268		.852	.790		.050
.075	1.224			.798	.732		-.1273			.795	.741		.075
.100	1.237		-.1096	.683	.683	.642	-.1253	-.1257	-.1089	.620	.602	.667	.100
.150	1.035		-.1086	.590	.550		-.1126	-.1240	-.1067	.529	.524	.483	.150
.250	-.746		-.1078	.515	.512	.473	-.737	-.1213	-.1074	.457	.450	.416	.250
.350	-.697	-.1155		.431	.442	.405	-.646	-.1157	-.1055	.404	.391	.348	.350
.450	-.684	-.1106		.375	.381	.340	-.646			.337	.353	.276	.450
.550	-.712	-.1061	-.1034	.305	.317	.266	-.682	-.1028	-.1024	.266	.260	.223	.550
.650	-.746	-.1007	-.1018	.238	.243	.212		-.957	-.1009			.187	.650
.750	-.770	-.944	-.995	.156	.175	.108	-.705	-.887	-.964				.750
.825			-.1012	.117	.133	.179			-.990				.825
.850	-.747	-.896		.156	.175		-.705			.204	.192		.850
.900	-.712	-.868		.117	.133	.108	-.697	-.809		.149	.147	.117	.900
.950	-.650	-.838					-.651	-.781					.950

TABLE II.- Continued  
PRESSURE COEFFICIENTS FOR WING IN THE PRESENCE OF THE BODIES

M = 1.05

Basic							Elliptical						
x/c	Upper surface			Lower surface			Upper surface			Lower surface			x/c
	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	
α = 0°													
.050	.014	.023		-.008	.022		-.050	-.047		-.046	.050		.050
.075	.009			.022	.029		-.058			-.037	.008		.075
.100		.027			.003			-.023	.092	-.011	.008		.100
.150	.009	.002	-.047			-.071	-.025	-.034	-.011	-.013	.061		.150
.250	-.052	-.054	-.014	-.034	-.012		-.018	-.061	-.025	-.015	-.001		.250
.350	-.047	-.090	-.057	-.023	-.041	-.070	.019	-.051	-.073	-.026	-.016	-.062	.350
.450	-.080	-.086	-.105	-.063	-.072	-.090	-.107	-.102	-.131	-.032	-.042	-.076	.450
.550	-.100	-.128	-.137	-.125	-.114	-.115	-.097	-.137	-.167	-.057	-.075	-.101	.550
.650	-.126	-.148	-.164	-.157	-.126	-.148	-.126	-.170	-.182	-.086	-.129	-.146	.650
.750	-.164		-.191	-.165	-.154	-.181			-.193	-.113		-.160	.750
.825			-.195			-.194						-.171	.825
.850	-.194	-.194		-.186	-.196		-.140			-.130	-.160		.850
.900	-.212	-.198		-.200		-.203	-.158	-.185		-.136		-.179	.900
.950	-.197	-.188					-.190	-.175					.950
α = 4°													
.050	-.178	-.608		.245	.278		-.252	-.612		.217	.304		.050
.075	-.180			.227	.260		-.254			.196	.273		.075
.100		-.090	-.579		.239			-.229	-.590		.233	.289	.100
.150	-.154	-.191	-.571		.163		-.204	-.227	-.586		.233		.150
.250	-.173	-.201	-.282	.150	.117	.232	-.150	-.208	-.322	.180	.184		.250
.350	-.183	-.241	-.248	.122	.117	.109	-.105	-.212	-.256	.139	.141	.123	.350
.450	-.204	-.221	-.275	.073	.072	.065	-.173			.109	.099	.077	.450
.550	-.230	-.260	-.293	.021	.019	.009	-.233	-.236	-.275	.076	.057	.024	.550
.650	-.254	-.279	-.307	-.021	-.005	-.044	-.228	-.270	-.297	.027		-.046	.650
.750	-.289		-.284	-.056	-.057	-.103	-.253	-.297	-.294	-.012	-.030	-.085	.750
.825			-.274			-.127			-.267			-.108	.825
.850	-.313	-.322		-.104	-.106		-.260			-.042	-.077		.850
.900	-.314	-.324		-.124		-.147	-.267	-.314		-.057		-.132	.900
.950	-.272	-.290					-.286	-.288					.950
α = 8°													
.050	-.553	-.977		.481	.485		-.568	-.974		.473	.494		.050
.075	-.355			.440	.441		-.457			.438	.449		.075
.100	-.347	-.818	-.938	.485	.400			-.856	-.903		.449		.100
.150	-.314	-.536	-.895	.322	.318	.425	-.362	-.599	-.869		.404	.438	.150
.250	-.286	-.375	-.829	.261	.257		-.274	-.394	-.815	.336	.329		.250
.350	-.291	-.373	-.800	.261	.257	.245	-.209	-.352	-.784	.269	.268	.249	.350
.450		-.368		.191	.202	.188	-.275			.225	.212	.192	.450
.550	-.351	-.400	-.507	.148	.155		-.336	-.362	-.499	.181	.163	.133	.550
.650	-.385	-.420	-.482	.100	.140	.057	-.377	-.389	-.472	.128	.106	.063	.650
.750	-.446	-.443	-.472	.051	.048	.017	-.375	-.414	-.445	.078	.063	.026	.750
.825			-.452			-.006			-.423			.003	.825
.850	-.439	-.474		.001	.005		-.387			.058	.026		.850
.900	-.458			-.015		-.033	-.388	-.436		.029		-.023	.900
.950	-.401	-.456					-.397	-.428					.950
α = 12°													
.050	-.973	1.059		.643	.619		-.930	1.025		.637	.624		.050
.075	-.565			.593	.569		-.698			.590	.577		.075
.100	-.545	-.994	1.142	.593	.522	.549	-.613	-.962	1.045		.533	.561	.100
.150	-.483	-.921	1.061	.444	.433		-.521	-.902	1.003	.456	.443	.428	.150
.250	-.396	-.823	1.008	.369	.367		-.391	-.802	-.971	.379	.377	.353	.250
.350	-.390	-.767	-.970	.291	.287		-.325	-.712	-.943	.324	.315	.290	.350
.450		-.645		.250	.252		-.364			.276	.263	.229	.450
.550	-.445	-.521	-.727	.194	.175	.157	-.429	-.531	-.745	.222	.157	.161	.550
.650	-.490	-.486	-.702	.142	.141	.114	-.491	-.503	-.499	.167		.123	.650
.750	-.551	-.514	-.675	.090	.096	.092	-.429	-.518	-.623			.099	.750
.825			-.642							.138	.116		.825
.850	-.543	-.521		.066	.096		-.490			.098		.062	.850
.900	-.557					.057	-.492	-.536					.900
.950	-.477	-.559					-.462	-.537					.950
α = 20°													
.050	1.187			.831	.814		1.203	1.182		.885	.824		.050
.075	1.179			.766	.766		1.194			.830	.779		.075
.100	1.158			.719	.719	.678	1.177	1.167	1.023		.736	.705	.100
.150	-.901	1.188	1.035	.644	.631	.588	1.025	1.149	1.006	.658	.642		.150
.250	-.672	1.127	1.009	.555	.553	.516	-.650	1.117	1.009	.570	.568	.528	.250
.350	-.624	1.091		.474	.486	.451	-.565	1.067	-.991	.502	.497	.463	.350
.450	-.623	1.035		.422	.429	.389	-.568			.451	.439	.399	.450
.550	-.661	-.967	-.965	.357	.368	.318	-.603	-.952	-.956	.387	.403	.327	.550
.650	-.686	-.887	-.944	.290	.298	.266	-.639	-.939	-.834	.319	.313	.277	.650
.750	-.674	-.814	-.924	.212	.231	.234			-.893	.262	.250	.245	.750
.825			-.929	.178	.190	.169	-.619		-.903	.211	.210	.180	.825
.850	-.661	-.774					-.602	-.694					.850
.900	-.626	-.735					-.545	-.674					.900
.950	-.568	-.731											.950

TABLE II.- Concluded  
PRESSURE COEFFICIENTS FOR WING IN THE PRESENCE OF THE BODIES

M = 1.115

Basic							Elliptical						
X/c	Upper surface			Lower surface			Upper surface			Lower surface			X/c
	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	0.20b/2	0.40b/2	0.70b/2	
$\alpha = 0^\circ$													
.050	.015	.076		-.012	.095		-.042	.027		-.040	.049		.050
.075	-.002			.000	.076		-.048			-.040	.057		.075
.100		.032			.000			.097					.100
.150	-.016	.009	-.002		.008	-.015	-.038	-.029	-.010		-.016		.150
.250	.001	-.037	-.010	-.002	.001	-.010	.011	-.051	.000	.007	-.008		.250
.350	-.013	-.046	-.055	-.009	-.013	-.046	.049	-.030	-.039	-.010	-.006	-.051	.350
.450		-.033	-.020	-.043	-.033	-.056	-.008			-.014	-.010	-.061	.450
.550	-.065	-.075	-.074	-.075	-.055		-.061	-.043	-.073	-.035	-.034	-.073	.550
.650	-.065	-.096	-.096	-.099		-.104	-.053	-.081	-.091	-.056		-.105	.650
.750	-.105	-.126	-.128	-.113	-.106	-.123	-.078	-.106	-.119	-.071	-.075	-.117	.750
.825			-.135			-.129			-.133			-.125	.825
.850	-.143			-.124	-.121		-.081			-.081	-.093		.850
.900	-.156			-.141		-.136	-.090	-.126		-.075		-.124	.900
.950	-.162	-.142					-.123	-.126					.950
$\alpha = 4^\circ$													
.050	-.184	-.462		.186	.282		-.224	-.409		.143	.273		.050
.075	-.176			.178	.258		-.222			.123	.251		.075
.100		-.170	-.378		.240			-.263	-.370		.227		.100
.150	-.162	-.158	-.433	.169	.193	.269	-.179	-.200	-.383		.203	.303	.150
.250	-.126	-.149	-.283	.154	.156	.196	-.131	-.171	-.344	.184	.207		.250
.350	-.135	-.179	-.213	.119	.119	.151	-.074	-.184	-.228	.163	.175	.157	.350
.450		-.171	-.172	.069	.087	.119	-.117			.145	.141	.125	.450
.550	-.185	-.202	-.228	.036	.087		-.177	-.180	-.215	.116	.106	.086	.550
.650	-.180	-.224	-.237	.036	.087		-.174	-.203	-.218	.079		.025	.650
.750	-.223	-.244	-.232	.013	.008	-.024	-.195	-.224	-.211	.053	.042	-.014	.750
.825			-.219	-.015	-.019				-.209			-.036	.825
.850	-.253			-.035		-.070	-.200			.034	.005		.850
.900	-.267			-.035			-.203	-.245		.028		-.058	.900
.950	-.256	-.254					-.223	-.237					.950
$\alpha = 8^\circ$													
.050	-.461	-.759		.459	.499		-.449	-.749		.429	.504		.050
.075	-.351			.431	.459		-.389			.418	.459		.075
.100	-.308	-.666	-.738	.431	.459			-.651	-.705		.422		.100
.150	-.267	-.559	-.690	.419	.419	.464	-.306	-.572	-.684	.418	.422	.479	.150
.250	-.236	-.322	-.648	.344	.353	.362	-.242	-.355	-.634	.354	.361		.250
.350	-.230	-.292	-.631	.296	.302	.304	-.172	-.306	-.616	.306	.314	.306	.350
.450		-.277		.241	.255	.254	-.204			.273	.268	.256	.450
.550	-.271	-.305	-.472	.208	.216		-.257	-.279	-.510	.238	.226	.204	.550
.650	-.302	-.328	-.395	.161	.186	.132	-.303	-.300	-.406	.192		.137	.650
.750	-.359	-.348	-.375	.125	.123	.090	-.298	-.324	-.356	.150	.138	.098	.750
.825			-.368			.067			-.356			.075	.825
.850	-.351	-.363		.077	.083		-.304			.133	.096		.850
.900	-.369			.061		.038	-.303	-.343		.113		.047	.900
.950	-.339	-.368					-.308	-.341					.950
$\alpha = 12^\circ$													
.050	-.798	-.962		.689	.659		-.747	-.934		.673	.666		.050
.075	-.501			.613	.570		-.627			.633	.620		.075
.100	-.424	-.865	-.916	.642	.613		-.598	-.840	-.892		.581		.100
.150	-.393	-.741	-.864	.570	.570	.603	-.454	-.806	-.858	.633	.620	.617	.150
.250	-.311	-.536	-.825	.492	.488	.481	-.305	-.547	-.808	.507	.498	.484	.250
.350	-.305	-.473	-.804	.426	.426	.415	-.247	-.478	-.787	.436	.436	.421	.350
.450		-.453		.356	.370	.359	-.271			.387	.381	.364	.450
.550	-.353	-.469	-.772	.318	.324		-.330	-.378	-.742	.346	.336	.307	.550
.650	-.390	-.473	-.621	.266	.254	.236		-.392	-.612	.297		.243	.650
.750	-.439	-.469	-.564	.222	.220	.196	-.380	-.417	-.532	.248	.237	.207	.750
.825			-.544			.172			-.493			.182	.825
.850	-.435	-.451		.174	.179		-.389			.226	.201		.850
.900	-.441			.155		.139	-.388	-.436		.193		.151	.900
.950	-.373	-.466					-.357	-.436					.950
$\alpha = 16.6^\circ$													
.050	-.947	-.990		.864	.810		-.1048	-.1015		.953	.895		.050
.075	-.719	-.964	-.1022	.812	.763		-.1035			.898	.850		.075
.100	-.596	-.975	-.1010	.627	.718	.714	-.1009	-.999	-.878		.811	.779	.100
.150	-.429	-.862	-.959	.641	.627	.601	-.843	-.986	-.868	.730	.722		.150
.250	-.397	-.842		.557	.558	.534	-.499	-.950	-.856	.651	.649	.614	.250
.350	-.409	-.815		.479	.499	.476	-.426			.584	.584	.554	.350
.450	-.455	-.734	-.774	.439	.447	.420	-.465	-.807	-.806	.538	.531	.492	.450
.550	-.497	-.589	-.751	.388	.395	.356		-.676	-.785	.480	.504	.428	.550
.650	-.530	-.539	-.737	.332	.337	.311	-.514	-.580	-.752	.417	.412	.380	.650
.750			-.713			.286			-.751			.350	.750
.825	-.508	-.538		.272	.287		-.495			.365	.353		.825
.850	-.472	-.542		.249	.317	.240	-.476	-.560		.318	.327	.288	.850
.900	-.432	-.546					-.429	-.540					.900
.950													.950

TABLE III  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.80

Basic						Elliptical					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E	Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$						$\alpha = 0^\circ$					
.055	.058		.050		.057	.055		.058		.074	.055
.166	-.003		-.011	-.006	-.001	-.005	.000	-.009	.001	.004	.166
.277	-.035	-.036	-.031	-.031	-.027	-.026	-.028	-.029	-.026	-.023	.277
.353			-.023					-.047			.353
.367	-.041	-.028	-.025	-.018	-.021	-.019	-.047	-.089	-.033		.367
.387		-.030		-.009	-.017	-.069	-.055		-.056		.387
.415	-.002					-.070					.415
.443	-.017	-.049		-.020	-.017	-.035	-.044		-.022	-.018	.443
.498	-.038	-.068		-.039	-.032	-.054	-.052		-.021	-.035	.498
.553	-.063	-.086		-.067	-.054	-.066	-.067		-.054	-.056	.553
.581	-.070					-.062					.581
.609	-.064	-.055		-.059	-.052	-.056	-.053		-.049	-.050	.609
.636	-.050					-.048					.636
.664	-.032	-.024		-.031	-.028	-.047	-.049		-.048	-.047	.664
.692	-.026		-.012			-.055		-.031			.692
.719	-.034	-.029	-.025	-.017	-.019	-.048	-.048	-.059	-.056	-.055	.719
.775	-.021			-.019	-.016	-.037	-.032	-.042	-.031		.775
.830	-.010	.000	-.018	-.014	-.016	-.018	-.017	-.033	-.033	-.028	.830
.871	-.023	-.010	-.006	.000	-.009	-.042	-.017	.003	.000	-.003	.871
.954	.055		.034		.033	.059		.039		.032	.954
$\alpha = 4^\circ$						$\alpha = 4^\circ$					
.055	.030		.058		.114	.035		.051		.114	.055
.166	-.018		-.016	.021	.042	-.017	-.020	-.015	.020	.042	.166
.277	-.035	-.040	-.034	-.005	.013	-.043	-.046	-.034	-.007	.012	.277
.353			-.026					-.047			.353
.367	-.044	-.038	-.022	.021	.035	-.057	-.073	-.095	.001		.367
.387		-.055		.048	.054	-.107	-.085		-.017		.387
.415	-.005					-.162					.415
.443	-.123	-.162		.105	.089	-.139	-.219		.102	.084	.443
.498	-.134	-.155		.074	.071	-.138	-.151		.085	.063	.498
.553	-.122	-.135		.019	.028	-.121	-.142		.022	.019	.553
.581	-.110					-.106					.581
.609	-.079	-.080		-.006	-.001	-.079	-.092		-.002	.001	.609
.636	-.055					-.062					.636
.664	-.030	-.028		-.007	.000	-.057	-.064		-.023	-.019	.664
.692	-.026		.004			-.057		-.021			.692
.719	-.029	-.025	-.011	-.005	-.004	-.055	-.045	-.049	-.045	-.038	.719
.775	-.016			-.009	-.005	-.026	-.028	-.035	-.022		.775
.830	.002	.014	-.009	-.008	-.011	-.003	-.009	-.025	-.022	-.018	.830
.871	-.008	.001	.005	.010	-.007	-.031	.001	.017	.007	-.001	.871
.954	.072		.042		.038	.059		.039		.037	.954
$\alpha = 8^\circ$						$\alpha = 8^\circ$					
.055	-.000		.003		.171	.004		.005		.178	.055
.166	-.032		-.061	.028	.089	-.030	-.048	-.059	.027	.093	.166
.277	-.038	-.062	-.073	.003	.062	-.037	-.062	-.072	.002	.064	.277
.353			-.059					-.079			.353
.367	-.056	-.066	-.052	.039	.095	-.051	-.081	-.116	.026		.367
.387		-.100		.087	.121	-.143	-.114		.020		.387
.415	-.005					-.258					.415
.443	-.255	-.415		.207	.179	-.298	-.454		.209	.175	.443
.498	-.228	-.296		.152	.151	-.261	-.279		.173	.138	.498
.553	-.208	-.254		.074	.086	-.202	-.228		.084	.077	.553
.581	-.181					-.168					.581
.609	-.136	-.160		.021	.029	-.127	-.147		.029	.032	.609
.636	-.092					-.096					.636
.664	-.045	-.066		-.007	.005	-.072	-.080		-.018	-.011	.664
.692	-.040		-.009			-.068		-.028			.692
.719	-.036	-.036	-.018	-.012	-.006	-.050	-.048	-.048	-.039	-.036	.719
.775	-.016			-.011	-.007	-.023	-.023	-.027	-.018		.775
.830	-.002	.009	-.011	-.007	-.011	-.003	-.006	-.021	-.022	-.017	.830
.871	-.018	.001	-.003	.008	-.003	-.045	.000	.018	.009	-.004	.871
.954	.080		.037		.030	.071		.043		.030	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.80

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	-.008		-.054		.250
.166	-.025		-.117	.040	.158
.277	-.014	-.066	-.119	.021	.132
.353			-.089		
.367	-.032	-.052	-.074	.076	.177
.387		-.095		.141	.213
.415	.004				
.443	-.274	-.485		.304	.277
.498	-.318	-.462		.244	.238
.553	-.303	-.391		.139	.153
.581	-.277				
.609	-.223	-.279		.047	.061
.636	-.166				
.664	-.100	-.161		-.030	.003
.692	-.077		-.062		
.719	-.055	-.082	-.032	-.026	-.009
.775	-.011			-.002	.005
.830	.009	.025	.006	.011	.005
.871	-.013	.011	.005	.028	.014
.954	.085		.053		.039
$\alpha = 20^\circ$					
.055	-.046		-.286		.408
.166			-.335	.015	.300
.277	.001	-.086	-.288	.016	.279
.353			-.194		
.367	-.020	-.016	-.150	.122	.346
.387		-.032		.212	.379
.415	-.009				
.443	-.184	-.393		.451	.437
.498	-.336	-.511		.390	.379
.553	-.465	-.549		.256	.260
.581	-.496				
.609	-.496	-.598		.104	.118
.636	-.495				
.664	-.477	-.627		-.124	-.057
.692	-.468		-.573		
.719	-.406	-.541	-.475	-.340	-.195
.775	-.229			-.254	-.126
.830	-.106	-.110	-.114	-.154	-.063
.871	-.044	-.048	-.056	-.045	-.012
.954	.029		.025		.037

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 12^\circ$					
-.019		-.058		.243	.055
-.034	-.078	-.123	.035	.149	.166
-.024	-.071	-.127	.013	.121	.277
		-.113			.353
-.028	-.067		.053		.367
-.136	-.108	-.143	.057		.387
-.273					.415
-.347	-.529		.301	.263	.443
-.375	-.441		.257	.216	.498
-.314	-.367		.145	.136	.553
-.272					.581
-.221	-.283		.048	.059	.609
-.177					.636
-.140	-.186		-.046	-.023	.664
-.117		-.077			.692
-.072	-.097	-.062	-.059	-.041	.719
-.026	-.043	-.020	-.014		.775
.000	-.009	-.011	-.010	-.004	.830
.043	-.015	.025	.020	-.002	.871
.065		.042		.028	.954
$\alpha = 20^\circ$					
-.040		-.284		.409	.055
-.041	-.124	-.329	.016	.301	.166
-.003	-.086	-.285	.015	.276	.277
		-.203			.353
.017	-.015	-.192	.103		.367
-.094	-.029		.123		.387
-.193					.415
-.265	-.416		.465	.433	.443
-.446	-.535		.403	.367	.498
-.492	-.568		.259	.248	.553
-.499					.581
-.513	-.611		.106	.122	.609
-.512					.636
-.500	-.633		-.134	-.076	.664
-.496		-.588			.692
-.395	-.539	-.522	-.380	-.231	.719
-.221	-.257	-.258	-.251		.775
-.098	-.112	-.122	-.158	-.060	.830
-.066	-.058	-.048	-.058	-.030	.871
.030		.017		.036	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.90

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.079		.076		.079
.166	.007		.003	.009	.013
.277	-.018	-.023	-.024	-.019	-.018
.353			-.017		
.367	-.008	-.004		-.004	-.003
.387		-.000	-.009	.007	-.001
.415	-.002				
.443	-.001	-.007		-.003	-.001
.498	-.031	-.044		-.027	-.023
.553	-.074	-.085		-.075	-.062
.581	-.082				
.609	-.062	-.058		-.062	-.058
.636	-.041				
.664	-.017	-.012		-.018	-.018
.692	-.015		.004		
.719	-.028	-.022	-.014	-.008	-.013
.775	-.018			-.012	-.010
.830	-.000	.013	-.010	-.007	-.009
.871	-.012	.001	.006	.010	-.003
.954	.074		.044		.044
$\alpha = 4^\circ$					
.055	.043		.069		.123
.166	-.015		-.011	.026	.044
.277	-.031	-.038	-.034	-.005	.011
.353			-.021		
.367	-.020	-.017	-.011	.024	.039
.387		-.026		.055	.059
.415	-.000				
.443	-.132	-.162		.119	.103
.498	-.194	-.216		.083	.081
.553	-.183	-.197		.011	.025
.581	-.133				
.609	-.084	-.081		-.021	-.014
.636	-.051				
.664	-.022	-.019		-.009	-.004
.692	-.017		.009		
.719	-.028	-.025	-.011	-.007	-.008
.775	-.016	-.002		-.012	-.008
.830	.002	.012	-.009	-.011	-.014
.871	-.012	-.000	.003	-.008	-.005
.954	.073		.043		.040
$\alpha = 8^\circ$					
.055	.017		.022		.188
.166	-.020		-.051	.036	.101
.277	-.021	-.045	-.060	.010	.069
.353			-.031		
.367	.001	-.011	-.017	.056	.110
.387		-.020		.108	.143
.415	.002				
.443	-.211	-.365		.233	.210
.498	-.366	-.422		.178	.177
.553	-.425	-.443		.086	.100
.581	-.375				
.609	-.105	-.156		.015	.027
.636	-.063				
.664	-.020	-.052		-.016	.000
.692	-.016		-.002		
.719	-.022	-.026	-.010	-.009	-.006
.775	-.009			-.008	-.002
.830	.003	.013	-.006	-.002	-.008
.871	-.021	.002	.001	.013	.002
.954	.086		.041		.040

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.071		.073		.089	.055
.006	.008	-.001	.009	.011	.166
-.023	-.024	-.027	-.024	-.021	.277
		-.048			.353
-.015	-.050	-.100	-.032		.367
-.063	-.049		-.054		.387
-.067					.415
-.022	-.036		-.014	-.010	.443
-.050	-.047		-.015	-.031	.498
-.080	-.084		-.065	-.066	.553
-.076					.581
-.063	-.064		-.059	-.057	.609
-.050					.636
-.043	-.048		-.044	-.042	.664
-.053		-.028			.692
-.052	-.051	-.061	-.058	-.057	.719
-.036	-.033	-.041	-.031		.775
-.015	-.016	-.033	-.033	-.028	.830
-.039	-.011	.008	.005	.002	.871
.065		.048		.039	.954
$\alpha = 4^\circ$					
.044		.061		.124	.055
-.016	-.014	-.012	.023	.043	.166
-.037	-.042	-.035	-.008	.010	.277
		-.046			.353
-.028	-.056	-.096	-.002		.367
-.082	-.051		-.019		.387
-.147					.415
-.159	-.219		.116	.097	.443
-.181	-.181		.096	.075	.498
-.176	-.194		.022	.019	.553
-.129					.581
-.082	-.099		-.014	-.009	.609
-.055					.636
-.044	-.045		-.028	-.024	.664
-.050		-.022			.692
-.047	-.046	-.055	-.051	-.045	.719
-.030	-.031	-.039	-.028		.775
-.009	-.013	-.034	-.031	-.026	.830
-.029	-.004	.014	.005	-.002	.871
.065		.040		.039	.954
$\alpha = 8^\circ$					
.013		.015		.186	.055
-.027	-.042	-.057	.028	.094	.166
-.030	-.054	-.068	.003	.061	.277
		-.059			.353
-.005	-.036	-.099	.033		.367
-.075	-.040		.028		.387
-.180					.415
-.287	-.424		.229	.197	.443
-.415	-.414		.190	.157	.498
-.400	-.408		.094	.084	.553
-.332					.581
-.110	-.148		.020	.026	.609
-.077					.636
-.055	-.068		-.031	-.022	.664
-.056		-.025			.692
-.044	-.046	-.047	-.046	-.044	.719
-.023	-.027	-.029	-.022		.775
-.006	-.013	-.025	-.027	-.024	.830
-.051	-.009	.017	.008	-.006	.871
.070		.038		.032	.954



TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.90

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	-.004		-.046		.257
.166	-.023		-.115	.042	.158
.277	-.001	-.052	-.111	.023	.132
.353			-.061		
.367	.026	.003	-.036	.089	.187
.387		-.017		.158	.227
.415	-.001				
.443	-.231	-.476		.331	.302
.498	-.432	-.522		.269	.262
.553	-.500	-.540		.159	.171
.581	-.509				
.609	-.449	-.438		.052	.063
.636	-.168				
.664	-.107	-.192		-.080	-.041
.692	-.100		-.121		
.719	-.080	-.112	-.086	-.082	-.067
.775	-.019			-.023	-.016
.830	.005	.008	.006	.005	-.003
.871	-.019	-.006	-.002	.026	.014
.954	.084		.048		.040
$\alpha = 20^\circ$					
.055	-.033		-.265		.423
.166			-.321	.031	.310
.277	.026	-.063	-.262	.035	.291
.353			-.145		
.367	.029	.033	-.100	.151	.370
.387		.017		.245	.410
.415	-.008				
.443	-.143	-.385		.499	.482
.498	-.404	-.525		.445	.433
.553	-.543	-.565		.318	.323
.581	-.573				
.609	-.570	-.610		.176	.188
.636	-.565				
.664	-.584	-.628		-.026	.019
.692	-.558		-.575		
.719	-.533	-.591	-.536	-.445	-.266
.775	-.360	-.008	-.008	-.497	-.398
.830	-.214	-.230	-.222	-.173	-.075
.871	-.127	-.095	-.076	-.089	-.036
.954	.008		-.000		.014

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 12^\circ$					
-.010		-.046		.254	.055
-.027	-.071	-.113	.038	.154	.166
-.010	-.058	-.113	.018	.123	.277
		-.079			.353
.032	-.012	-.107	.070		.367
-.053	-.022		.073		.387
-.191					.415
-.330	-.527		.331	.293	.443
-.467	-.528		.283	.246	.498
-.465	-.506		.167	.158	.553
-.486					.581
-.449	-.480		.057	.066	.609
-.161					.636
-.173	-.227		-.083	-.058	.664
-.133		-.132			.692
-.078	-.118	-.100	-.104	-.091	.719
-.025	-.053	-.027	-.025		.775
.004	-.018	-.010	-.012	-.011	.830
-.049	-.026	.028	.025	.002	.871
.064		.044		.034	.954
$\alpha = 20^\circ$					
-.023		-.257		.427	.055
-.027	-.110	-.314	.038	.312	.166
.027	-.058	-.256	.037	.293	.277
		-.153			.353
.071	.038	-.137	.136		.367
-.042	.027		.156		.387
-.168					.415
-.262	-.409		.518	.483	.443
-.454	-.571		.458	.428	.498
-.528	-.582		.324	.316	.553
-.543					.581
-.541	-.621		.183	.196	.609
-.546					.636
-.549	-.641		-.030	.008	.664
-.564		-.595			.692
-.463	-.585	-.572	-.482	-.294	.719
-.325	-.441	-.436	-.490		.775
-.190	-.219	-.204	-.171	-.067	.830
-.157	-.120	-.093	-.090	-.058	.871
.022		.011		.021	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.94

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.081		.077		.082
.166	.006		-.001	.006	.010
.277	-.028	-.030	-.029	-.027	-.025
.353			-.021		
.367	-.011	-.008	-.009	-.005	-.004
.387		-.003		.007	-.001
.415	-.002				
.443	.002	-.015		-.002	.002
.498	-.035	-.055		-.027	-.024
.553	-.100	-.118		-.099	-.084
.581	-.129				
.609	-.110	-.103		-.105	-.096
.636	-.064				
.664	-.029	-.015		-.021	-.020
.692	-.016		.002		
.719	-.031	-.025	-.018	-.013	-.016
.775	-.023			-.017	-.016
.830	-.004	.007	-.016	-.012	-.016
.871	-.017	-.004	.003	.006	-.007
.954	.070		.043		.041
$\alpha = 4^\circ$					
.055	.049		.074		.130
.166	-.013		-.009	.026	.046
.277	-.031	-.038	-.034	-.007	.009
.353			-.016		
.367	-.006	-.004	-.001	.026	.041
.387		-.007		.059	.063
.415	-.002				
.443	-.110	-.142		.126	.111
.498	-.177	-.198		.088	.084
.553	-.252	-.266		.003	.016
.581	-.289				
.609	-.268	-.267		-.049	-.040
.636	-.039				
.664	.009	.008		-.006	-.003
.692	.004		.023		
.719	-.015	-.013	-.002	-.001	-.003
.775	-.014			-.012	-.009
.830	.001	.010	-.011	-.010	-.013
.871	-.012	-.001	.006	.008	-.003
.954	.074		.044		.043
$\alpha = 8^\circ$					
.055	.021		.026		.193
.166	-.020		-.049	.038	.099
.277	-.018	-.042	-.058	.011	.068
.353			-.020		
.367	.023	.012	-.003	.063	.115
.387		.012		.116	.149
.415	-.002				
.443	-.163	-.307		.245	.221
.498	-.325	-.376		.187	.185
.553	-.413	-.452		.089	.104
.581	-.457				
.609	-.478	-.508		.003	.014
.636	-.502				
.664	-.070	-.131		-.034	-.018
.692	.024		-.014		
.719	.036	.018	.018	.002	-.001
.775	.022			.005	.006
.830	.013	.022	-.000	.004	-.003
.871	-.019	.005	.000	.016	.005
.954	.090		.043		.047

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.077		.079		.094	.055
.004	.009	-.002	.008	.010	.166
-.026	-.029	-.030	-.028	-.025	.277
		-.050			.353
-.015	-.057	-.111	-.039		.367
-.065	-.050		-.058		.387
-.067					.415
-.018	-.030		-.010	-.008	.443
-.068	-.042		-.013	-.029	.498
-.103	-.107		-.082	-.082	.553
-.109					.581
-.083	-.084		-.077	-.076	.609
-.053					.636
-.043	-.043		-.042	-.043	.664
-.052		-.028			.692
-.056	-.054	-.067	-.064	-.061	.719
-.043	-.038	-.047	-.037		.775
-.020	-.022	-.039	-.040	-.035	.830
-.041	-.013	.007	.004	.000	.871
.066		.047		.040	.954
$\alpha = 4^\circ$					
.052		.068		.133	.055
-.013	-.014	-.011	.026	.046	.166
-.034	-.041	-.036	-.011	.008	.277
		-.044			.353
-.012	-.046	-.098	-.003		.367
-.061	-.030		-.015		.387
-.123					.415
-.155	-.212		.126	.106	.443
-.177	-.180		.106	.083	.498
-.235	-.243		.018	.017	.553
-.270					.581
-.225	-.238		-.032	-.025	.609
-.036					.636
-.017	-.017		-.026	-.024	.664
-.031		-.010			.692
-.039	-.038	-.052	-.050	-.046	.719
-.032	-.030	-.040	-.028		.775
-.013	-.016	-.036	-.034	-.030	.830
-.032	-.005	.014	.005	.000	.871
.065		.041		.040	.954
$\alpha = 8^\circ$					
.022		.024		.193	.055
-.023	-.039	-.052	.033	.098	.166
-.024	-.047	-.063	.006	.064	.277
		-.045			.353
.023	-.010	-.086	.042		.367
-.034	-.001		.039		.387
-.129					.415
-.231	-.361		.245	.214	.443
-.372	-.378		.201	.170	.498
-.391	-.412		.101	.092	.553
-.424					.581
-.449	-.465		.013	.020	.609
-.466					.636
-.193	-.153		-.044	-.035	.664
.023		-.007			.692
.029	.008	-.008	-.030	-.032	.719
.008	-.002	-.010	-.005		.775
.006	-.004	-.017	-.018	-.017	.830
-.049	-.009	.020	.013	.003	.871
.073		.038		.041	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.94

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	.008		-.036		.265
.166	-.016		-.107	.047	.163
.277	.011	-.038	-.099	.030	.138
.353			-.037		
.367	.059	.037	-.011	.106	.200
.387		.025		.177	.243
.415	-.001				
.443	-.178	-.418		.351	.323
.498	-.470	-.531		.290	.284
.553	-.552	-.568		.181	.195
.581	-.571				
.609	-.569	-.586		.075	.087
.636	-.564				
.664	-.545	-.416		-.064	-.029
.692	-.032		-.189		
.719	.009	-.057	-.102	-.161	-.166
.775	.021		-.011	-.011	-.008
.830	.022	.014	.015	.018	.007
.871	-.013	-.000	-.001	.034	.021
.954	.090		.051		.051
$\alpha = 20^\circ$					
.055	-.013		-.244		.444
.166			-.310	.050	.327
.277	.051	-.039	-.241	.056	.298
.353			-.114		
.367	.061	.063	-.068	.176	.395
.387		.037		.273	.444
.415	.002				
.443	-.189	-.428		.535	.523
.498	-.511	-.683		.483	.476
.553	-.625	-.733		.364	.372
.581	-.665				
.609	-.662	-.773		.231	.245
.636	-.660				
.664	-.666	-.808		.043	.089
.692	-.670		-.541		
.719	-.669	-.631	-.539	-.385	-.188
.775	-.134			-.545	-.408
.830	-.070	-.153	-.177	-.317	-.425
.871	-.057	-.044	-.002	.013	.035
.954	.115		.089		.095

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 12^\circ$					
.002		-.034		.267	.055
-.019	-.060	-.106	.048	.160	.166
.004	-.043	-.102	.028	.131	.277
		-.057			.353
.071	.025	-.083	.088		.367
-.006	.022		.093		.387
-.133					.415
-.269	-.465		.355	.317	.443
-.494	-.531		.306	.272	.498
-.508	-.534		.192	.184	.553
-.536					.581
-.570	-.630		.084	.092	.609
-.608					.636
-.620	-.431		-.065	-.041	.664
-.105		-.204			.692
.023	-.049	-.045	-.082	-.077	.719
.049	.019	.010	.001		.775
.031	.012	-.002	-.003	-.002	.830
-.036	-.008	.029	.028	.012	.871
.073		.043		.038	.954
$\alpha = 20^\circ$					
-.011		-.242		.441	.055
-.018	-.099	-.308	.047	.322	.166
.041	-.043	-.244	.051	.305	.277
		-.133			.353
.095	.059	-.115	.154		.367
-.020	.039		.171		.387
-.183					.415
-.296	-.474		.545	.511	.443
-.529	-.708		.491	.463	.498
-.604	-.721		.361	.354	.553
-.628					.581
-.629	-.778		.226	.238	.609
-.632					.636
-.639	-.787		.029	.064	.664
-.659		-.537			.692
-.629	-.621	-.571	-.420	-.222	.719
-.095	-.362	-.396	-.531		.775
-.061	-.155	-.171	-.307	-.383	.830
-.114	-.045	.025	.020	.022	.871
.094		.086		.082	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.98

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.093		.091		.093
.166	.009		.004	.010	.012
.277	-.027	-.031	-.033	-.029	-.028
.353			-.020		
.367	.006	.005	.002	.004	.006
.387		.011		.017	.009
.415	-.004				
.443	.024	.020		.015	.020
.498	-.009	-.019		-.004	-.001
.553	-.082	-.090		-.077	-.064
.581	-.120				
.609	-.145	-.149		-.149	-.144
.636	-.178				
.664	-.214	-.202		-.225	-.211
.692	-.066		-.033		
.719	-.005	-.007	-.006	.007	.011
.775	.003			.008	.011
.830	.012	.023	.001	.003	.001
.871	-.005	.008	.014	.016	.006
.954	.081		.052		.049
$\alpha = 4^\circ$					
.055	.060		.086		.139
.166	-.010		-.006	.029	.048
.277	-.030	-.038	-.036	-.009	.008
.353			-.005		
.367	.017	.017	.017	.040	.054
.387		.021		.075	.079
.415	-.003				
.443	-.075	-.107		.151	.137
.498	-.147	-.163		.116	.111
.553	-.216	-.230		.027	.038
.581	-.252				
.609	-.271	-.286		-.062	-.055
.636	-.297				
.664	-.332	-.334		-.170	-.148
.692	-.180		-.095		
.719	-.012	-.027	-.047	-.068	-.079
.775	.021			.013	.014
.830	.030	.038	.019	.016	.013
.871	.011	.020	.026	.029	.018
.954	.085		.059		.057
$\alpha = 8^\circ$					
.055	.032		.038		.204
.166	-.016		-.045	.044	.104
.277	-.008	-.034	-.052	.017	.074
.353			.002		
.367	.058	.046	.025	.084	.134
.387		.054		.142	.174
.415	-.004				
.443	-.107	-.248		.276	.251
.498	-.276	-.324		.219	.218
.553	-.363	-.397		.120	.136
.581	-.404				
.609	-.425	-.452		.025	.035
.636	-.447				
.664	-.483	-.500		-.083	-.055
.692	-.496		-.187		
.719	-.101	-.160	-.169	-.176	-.181
.775	-.019			-.127	-.156
.830	.029	.025	.005	.012	.009
.871	.017	.032	-.004	.044	.037
.954	.119		.071		.081

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.092		.092		.108	.055
.010	.015	.004	.014	.017	.166
-.028	-.029	-.031	-.029	-.026	.277
		-.054			.353
-.018	-.065	-.119	-.047		.367
-.052	-.039		-.051		.387
-.051					.415
.007	-.005		.011	.015	.443
-.018	-.010		.015	-.003	.498
-.082	-.085		-.065	-.065	.553
-.112					.581
-.141	-.156		-.134	-.129	.609
-.172					.636
-.191	-.183		-.162	-.175	.664
-.048		-.031			.692
-.022	-.025	-.043	-.039	-.031	.719
-.023	-.020	-.029	-.019		.775
-.011	-.013	-.033	-.031	-.028	.830
-.030	-.005	.015	.012	.011	.871
.077		.056		.050	.954
$\alpha = 4^\circ$					
.062		.080		.144	.055
-.010	-.008	-.008	.031	.051	.166
-.039	-.046	-.038	-.010	.007	.277
		-.037			.353
.001	-.036	-.095	.003		.367
-.032	-.002		-.003		.387
-.087					.415
-.121	-.176		.151	.135	.443
-.158	-.160		.132	.112	.498
-.200	-.205		.041	.041	.553
-.234					.581
-.260	-.282		-.052	-.042	.609
-.287					.636
-.315	-.328		-.148	-.134	.664
-.192		-.096			.692
-.033	-.046	-.086	-.113	-.113	.719
.003	-.006	-.017	-.009		.775
.013	.006	-.013	-.013	-.010	.830
-.014	.014	.031	.022	.020	.871
.075		.052		.057	.954
$\alpha = 8^\circ$					
.031		.034		.206	.055
-.017	-.033	-.047	.037	.100	.166
-.017	-.041	-.056	.010	.067	.277
		-.023			.353
.053	.020	-.066	.060		.367
.007	.039		.061		.387
-.077					.415
-.173	-.299		.274	.242	.443
-.318	-.324		.233	.197	.498
-.336	-.358		.130	.122	.553
-.371					.581
-.398	-.432		.031	.039	.609
-.425					.636
-.445	-.470		-.081	-.063	.664
-.496		-.212			.692
-.145	-.190	-.205	-.212	-.228	.719
-.030	-.059	-.099	-.152		.775
.028	.009	-.007	-.006	-.005	.830
-.005	.026	.046	.046	.040	.871
.105		.069		.076	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 0.96

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	.018		-.023		.277
.166	-.009		-.100	.055	.170
.277	.028	-.022	-.085	.043	.149
.353			-.007		
.367	.098	.074	.023	.131	.223
.387		.069		.203	.270
.415	-.003				
.443	-.119			.382	.354
.498	-.407	-.349		.322	.318
.553	-.489	-.463		.216	.230
.581	-.519	-.521			
.609	-.539	-.578		.115	.125
.636	-.559				
.664	-.603	-.643		-.015	.018
.692	-.624		-.307		
.719	-.417	-.308	-.236	-.218	-.210
.775	-.070			-.237	-.231
.830	-.014	-.018	-.087	-.124	-.177
.871	-.007	.009	-.002	.017	.010
.954	.122		.087		.090
$\alpha = 20^\circ$					
.055	-.007		-.229		.451
.166			-.284	.062	.331
.277	.071	-.021	-.221	.070	.320
.353			-.081		
.367	.097	.096	-.036	.196	.409
.387		.069		.296	.466
.415	-.004				
.443	-.259	-.516		.558	.547
.498	-.603	-.786		.513	.505
.553	-.726	-.706		.398	.408
.581	-.666				
.609	-.666	-.780		.271	.285
.636	-.649				
.664	-.677	-.806		.095	.143
.692	-.695		-.440		
.719	-.749	-.568	-.390	-.292	-.125
.775	-.298	-.004	-.005	-.451	-.282
.830	-.077	-.105	-.246	-.454	-.416
.871	-.192	-.160	-.160	-.176	-.305
.954	.071		.018		.031

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 12^\circ$					
.015		-.020		.279	.055
-.011	-.053	-.098	.054	.167	.166
.021	-.027	-.087	.039	.143	.277
		-.028			.353
.109	.064	-.051	.114		.367
.041	.069		.121		.387
-.073					.415
-.203	-.392			.350	.443
-.428	-.466		.386	.306	.498
-.446	-.470		.229	.223	.553
-.472					.581
-.506	-.564		.124	.134	.609
-.542					.636
-.562	-.607		-.016	.005	.664
-.610		-.319			.692
-.553	-.312	-.257	-.253	-.237	.719
-.088	-.135	-.206	-.243		.775
-.013	-.048	-.100	-.155	-.198	.830
-.027	-.010	.019	.021	.013	.871
.102		.081		.085	.954
$\alpha = 20^\circ$					
.000		-.224		.452	.055
.001	-.087	-.283	.062	.331	.166
.071	-.017	-.216	.073	.320	.277
		-.094			.353
.135	.097	-.074	.181		.367
.025	.079		.200		.387
-.161					.415
-.333	-.569			.544	.443
-.595	-.816		.578	.500	.498
-.698	-.694		.401	.397	.553
-.577					.581
-.631	-.738		.275	.289	.609
-.618					.636
-.617	-.745		.088	.124	.664
-.646		-.434			.692
-.678	-.552	-.419	-.316	-.151	.719
-.327	-.388	-.482	-.451		.775
-.069	-.132	-.281	-.493	-.384	.830
-.242	-.140	-.160	-.257	-.476	.871
.050		.008		.031	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 1.05

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.126		.122		.124
.166	.053		.047	.054	.057
.277	.012	.008	.007	.010	.010
.353			-.031		
.367	-.022	-.022	-.026	-.028	-.020
.387		.011		.014	.013
.415	-.004				
.443	.012	.005		.004	.009
.498	-.016	-.017		.008	.003
.553	-.058	-.069		-.048	-.033
.581	-.090				
.609	-.108	-.115		-.106	-.104
.636	-.132				
.664	-.165	-.173		-.183	-.157
.692	-.176		-.072		
.719	-.099	-.101	-.092	-.087	-.081
.775	-.087			-.073	-.070
.830	-.073	-.063	-.085	-.083	-.083
.871	-.050	-.037	-.031	-.030	-.040
.954	.076		.048		.042
$\alpha = 4^\circ$					
.055	.092		.116		.173
.166	.032		.035	.069	.087
.277	-.002	-.010	-.005	.024	.038
.353			-.043		
.367	-.021	-.033	-.042	-.022	-.001
.387		-.004		.026	.039
.415	-.004				
.443		-.088		.167	.156
.498	-.119	-.127		.150	.145
.553	-.172	-.191		.073	.087
.581	-.207				
.609	-.222	-.238		-.005	.003
.636	-.244				
.664	-.276	-.290		-.112	-.084
.692	-.272		-.093		
.719	-.093	-.108	-.098	-.105	-.115
.775	-.063		-.087	-.087	-.091
.830	-.044	-.042	-.067	-.072	-.076
.871	-.024	-.014	-.010	-.009	-.018
.954	.002		-.032		-.042
$\alpha = 8^\circ$					
.055	.063		.069		.235
.166	.016		-.017	.072	.132
.277	-.011	-.036	-.047	.023	.079
.353			-.065		
.367	.076	.068	.044	.083	.129
.387		.097		.171	.202
.415	-.007				
.443	-.039	-.175		.325	.300
.498	-.214	-.262		.272	.271
.553	-.296	-.327		.180	.195
.581	-.332				
.609	-.353	-.380		.089	.099
.636	-.373				
.664	-.404	-.428		-.018	.009
.692	-.426		-.153		
.719	-.170	-.160	-.133	-.125	-.141
.775	-.057		-.143	-.144	
.830	-.080	-.081	-.119	-.122	-.137
.871	-.107	-.076	-.005	-.056	-.067
.954	-.042		-.094		-.119

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.120		.121		.136	.055
.053	.056	.046	.055	.057	.166
.016	.011	.010	.012	.015	.277
		-.032			.353
-.015	-.057	-.095	-.039		.367
-.049	-.051		-.069		.387
-.129					.415
-.059	-.062		-.042	-.048	.443
-.040	-.020		.011	-.008	.498
-.058	-.057		-.040	-.040	.553
-.086					.581
-.108	-.123		-.097	-.091	.609
-.129					.636
-.156	-.172		-.164	-.152	.664
-.195		-.078			.692
-.092	-.107	-.128	-.125	-.106	.719
-.104	-.100	-.107	-.095		.775
-.080	-.081	-.105	-.106	-.105	.830
-.120	-.095	-.071	-.077	-.078	.871
.076		.056		.050	.954
$\alpha = 4^\circ$					
.089		.104		.171	.055
.029	.029	.030	.062	.085	.166
-.009	-.013	-.004	.023	.041	.277
		-.042			.353
-.031	-.076	-.102	-.018		.367
-.059	-.043		-.070		.387
-.115					.415
-.109	.163		.163	.150	.443
-.143	-.133		.165	.145	.498
-.164	-.169		.084	.086	.553
-.195					.581
-.215	-.238		.005	.013	.609
-.240					.636
-.263	-.279		-.093	-.074	.664
-.294		-.097			.692
-.106	-.117	-.134	-.141	-.141	.719
-.081	-.089	-.112	-.113		.775
-.068	-.073	-.099	-.100	-.099	.830
-.094	-.065	-.045	-.058	-.061	.871
.000		-.027		-.030	.954
$\alpha = 8^\circ$					
.064		.071		.239	.055
.016	.001	-.013	.067	.130	.166
-.015	-.040	-.054	.013	.071	.277
		-.090			.353
.032	-.013	-.122	-.005		.367
.045	.076		.078		.387
-.016					.415
-.104	-.223		.323	.288	.443
-.251	-.260		.284	.251	.498
-.269	-.288		.185	.178	.553
-.299					.581
-.327	-.359		.090	.100	.609
-.353					.636
-.371	-.396		-.014	.002	.664
-.420		-.180			.692
-.204	-.167	-.158	-.163	-.178	.719
-.059	-.084	-.128	-.153		.775
-.073	-.091	-.119	-.130	-.133	.830
-.156	-.107	-.078	-.099	-.122	.871
-.039		-.082		-.108	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 1.03

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	.047		.008		.300
.166	.008		-.083	.070	.185
.277	.003	-.048	-.109	.027	.133
.353			.037		
.367	.153	.125	.072	.168	.258
.387		.128		.247	.313
.415	-.006				
.443	-.042	-.264		.431	.404
.498	-.325	-.379		.374	.370
.553	-.401	-.435		.271	.285
.581	-.432				
.609	-.454	-.492		.174	.183
.636	-.474				
.664	-.517	-.557		.053	.086
.692	-.536		-.255		
.719	-.516	-.265	-.187	-.166	-.135
.775	-.089			-.200	-.182
.830	-.124	-.117	-.187	-.205	-.217
.871	-.142	-.107	-.005	-.110	-.159
.954	-.094		-.144		-.197
$\alpha = 20^\circ$					
.055	.015		-.197		.472
.166			-.276	.073	.341
.277	.107	.013	-.204	.095	.342
.353			-.029		
.367	.150	.147	.019	.236	.443
.387		.129		.337	.503
.415	-.013				
.443	-.193	-.434		.597	.586
.498	-.521	-.688		.555	.548
.553	-.636	-.659		.444	.456
.581	-.617				
.609	-.614	-.681		.323	.338
.636	-.603				
.664	-.599	-.716		.158	.202
.692	-.599		-.388		
.719	-.652	-.495	-.325	-.229	-.054
.775	-.269			-.379	-.210
.830	-.170	-.172	-.316	-.438	-.351
.871	-.202	-.185	-.221	-.244	-.362
.954	-.222		-.203		-.263

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 12^\circ$					
.036		.010		.303	.055
.006	-.038	-.083	.070	.182	.166
-.017	-.066	-.117	.016	.122	.277
		.013			.353
.157	.112	-.010	.150		.367
.099	.122		.160		.387
-.008					.415
-.131	-.312		.431	.393	.443
-.348	-.390		.386	.346	.498
-.368	-.395		.277	.268	.553
-.394					.581
-.427	-.483		.178	.183	.609
-.461					.636
-.478	-.520		.048	.066	.664
-.524		-.268			.692
-.523	-.263	-.207	-.200	-.168	.719
-.080	-.130	-.196	-.207		.775
-.123	-.137	-.184	-.213	-.212	.830
-.218	-.159	-.132	-.184	-.268	.871
-.109		-.142		-.204	.954
$\alpha = 20^\circ$					
.026		-.187		.473	.055
.013	-.070	-.263	.076	.343	.166
.110	.018	-.198	.099	.344	.277
		-.037			.353
.193	.151	-.017	.223		.367
.090	.138		.244		.387
-.087					.415
-.254	-.476		.619	.587	.443
-.508	-.710		.570	.543	.498
-.608	-.626		.450	.449	.553
-.524					.581
-.578	-.667		.330	.343	.609
-.564					.636
-.569	-.652		.156	.189	.664
-.569		-.370			.692
-.568	-.475	-.343	-.246	-.074	.719
-.271	-.326	-.418	-.379		.775
-.167	-.190	-.309	-.445	-.316	.830
-.345	-.219	-.223	-.353	-.430	.871
-.215		-.204		-.282	.954

TABLE III.- Continued  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 1.115

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.100		.099		.112
.166	.030		.020	.029	.030
.277	-.002	-.004	-.002	-.000	.002
.353			-.026		
.367	-.039	-.035	-.040	-.030	-.021
.387		.014		.008	.002
.415					
.443	.006	.003		.008	-.005
.498	.031	-.001		.018	.012
.553	-.024	-.029		-.018	-.008
.581	-.047				
.609	-.069	-.076		-.062	-.061
.636	-.094				
.664	-.115	-.129		-.120	-.102
.692	-.132		-.067		
.719	-.117	-.077	-.057	-.070	-.107
.775	-.038			-.037	-.027
.830	-.062	-.050	-.069	-.054	-.058
.871	-.037	-.019		-.016	-.021
.954	.002		-.029		-.041
$\alpha = 4^\circ$					
.055	.071		.092		.156
.166	.008		.011	.051	.070
.277	-.017	-.021	-.012	.014	.030
.353			-.036		
.367	-.042	-.053	-.057	-.019	.010
.387		.002		.012	.028
.415					
.443	-.038	-.085		.123	.082
.498	-.086	-.103		.176	.177
.553	-.127	-.143		.113	.125
.581	-.157				
.609	-.175	-.189		.058	.063
.636	-.196				
.664	-.218	-.236		-.021	.005
.692	-.233		-.074		
.719	-.135	-.079	-.053	-.067	-.092
.775	-.018			-.066	-.059
.830	-.045	-.042	-.070	-.066	-.079
.871	-.035	-.022		-.024	-.027
.954	-.004		-.039		-.065
$\alpha = 8^\circ$					
.055	.038		.043		.213
.166	-.010		-.035	.054	.116
.277	-.025	-.048	-.058	.014	.072
.353			-.093		
.367	-.052	-.069	-.104	-.035	.013
.387		-.030		.007	.047
.415	-.010				
.443	.001	-.142		.321	.297
.498	-.163	-.210		.302	.303
.553	-.232	-.259		.229	.244
.581	-.261				
.609	-.284	-.301		.155	.165
.636	-.294				
.664	-.318	-.344		.057	.082
.692	-.330		-.122		
.719	-.269	-.124	-.080	-.095	-.057
.775	-.016			-.098	-.083
.830	-.049	-.053	-.097	-.107	-.126
.871	-.069	-.042	-.009	-.033	-.061
.954	.010		-.033		-.071

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.097		.096		.110	.055
.024		.016	.024	.025	.166
-.001	-.003	-.002	-.002	.001	.277
		-.030			.353
-.018	-.048	-.068	-.029		.367
-.026	-.048		-.054		.387
-.117					.415
-.061	-.055		-.035	-.054	.443
.001	-.010		.016	.000	.498
-.010	-.006		-.003	-.002	.553
-.034					.581
-.053	-.065		-.050	-.049	.609
-.078					.636
-.096	-.111		-.097	-.087	.664
-.138		-.071			.692
-.120	-.077	-.084	-.097	-.122	.719
-.050	-.056	-.070	-.051		.775
-.055	-.053	-.069	-.069	-.068	.830
-.104	-.077	-.052	-.062	-.065	.871
-.019		-.041		-.056	.954
$\alpha = 4^\circ$					
.072		.090		.157	.055
.005	.008	.007	.047	.067	.166
-.017	-.021	-.014	.014	.030	.277
		-.037			.353
-.020	-.064	-.077	-.015		.367
-.037	-.044		-.053		.387
-.107					.415
-.085	-.137			.000	.443
-.119	-.109		.086	.176	.498
-.113	-.114		.129	.129	.553
-.139					.581
-.159	-.171		.071	.076	.609
-.179					.636
-.193	-.207		-.007	.012	.664
-.230		-.080			.692
-.139	-.077	-.072	-.092	-.107	.719
-.027	-.043	-.081	-.080		.775
-.043	-.047	-.073	-.081	-.085	.830
-.089	-.070	-.053	-.073	-.085	.871
-.024		-.058		-.075	.954
$\alpha = 8^\circ$					
.034		.043		.209	.055
-.013	-.023	-.037	.049	.108	.166
-.024	-.046	-.057	.012	.069	.277
		-.092			.353
-.024	-.074	-.117	-.029		.367
-.051	-.049		-.066		.387
-.089					.415
-.058	-.182		.305	.278	.443
-.192	-.212		.317	.294	.498
-.204	-.216		.239	.241	.553
-.220					.581
-.250	-.273		.163	.173	.609
-.272					.636
-.283	-.305		.073	.088	.664
-.321		-.148			.692
-.280	-.142	-.110	-.130	-.077	.719
-.012	-.052	-.104	-.109		.775
-.037	-.057	-.090	-.114	-.123	.830
-.119	-.070	-.053	-.092	-.136	.871
-.023		-.053		-.086	.954



TABLE III.- Concluded  
PRESSURE COEFFICIENTS FOR BODIES IN THE PRESENCE OF THE WING

M = 1.115

Basic					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	.024		-.014		.280
.166	-.018		-.104	.056	.171
.277	-.021	-.073	-.122	.011	.116
.353			-.158		
.367	-.048	-.071		-.035	.059
.387		.040	-.169	.012	.096
.415	-.008				
.443	.041	-.183		.470	.436
.498	-.237	-.296		.427	.424
.553	-.306	-.360		.335	.354
.581	-.335				
.609	-.349	-.401		.248	.261
.636	-.367				
.664	-.400	-.465		.148	.177
.692	-.416		-.186		
.719	-.446	-.181	-.115	-.113	-.019
.775	.016			-.130	-.104
.830	-.071	-.067	-.138	-.136	-.138
.871	-.058	-.050	-.008	-.065	-.136
.954	-.048	-.054	-.054	-.097	
$\alpha = 16.6^\circ$					
.055	-.006		-.121		.382
.166			-.229	.058	.252
.277	-.024	-.115	-.262	-.000	.189
.353			-.280		
.367	.161	.055	-.135	-.059	.122
.387		.161		.269	.373
.415	-.008				
.443	.010	-.241		.602	.579
.498	-.327	-.411		.567	.556
.553	-.380	-.457		.461	.479
.581	-.444				
.609	-.448	-.502		.364	.379
.636	-.456				
.664	-.489	-.549		.232	.275
.692	-.510		-.252		
.719	-.558	-.296	-.132	-.114	.045
.775	-.006			-.187	-.084
.830	-.049	-.052	-.142	-.164	-.159
.871	-.094	-.053	-.070	-.067	-.145
.954	.059		-.039		-.145

Elliptical					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 12^\circ$					
.010		-.012		.279	.055
-.020	-.060	-.100	.060	.170	.166
-.021	-.070	-.120	.009	.111	.277
		-.153			.353
-.036	-.090	-.175	-.030		.367
-.034	.003		-.073		.387
.031					.415
-.040	-.219		.462	.428	.443
-.245	-.303		.426	.406	.498
-.269	-.310		.337	.344	.553
-.299					.581
-.324	-.380		.251	.265	.609
-.349					.636
-.365	-.411		.144	.166	.664
-.400		-.188			.692
-.422	-.189	-.132	-.135	-.046	.719
.013	-.064	-.128	-.139		.775
-.053	-.075	-.117	-.143	-.136	.830
-.138	-.105	-.071	-.135	-.212	.871
-.075		-.081		-.120	.954
$\alpha = 20^\circ$					
-.009		-.210		.453	.055
-.048	-.133	-.327	.057	.313	.166
-.055	-.141	-.348	-.014	.238	.277
		.038			.353
.282	.184	.045	.172		.367
.223	.222		.273		.387
.052					.415
-.107	-.327		.695	.657	.443
-.368	-.571		.647	.627	.498
-.471	-.506		.538	.541	.553
-.380					.581
-.437	-.536		.428	.443	.609
-.441					.636
-.435	-.514		.269	.303	.664
-.440		-.249			.692
-.427	-.340	-.214	-.119	.059	.719
-.168	-.205	-.306	-.251		.775
.016	-.052	-.189	-.333	-.183	.830
-.181	-.071	-.088	-.224	-.305	.871
-.098		-.077		-.126	.954

TABLE IV  
PRESSURE COEFFICIENTS FOR BASIC BODY ALONE

M = 0.80

Basic body alone					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.059		.064		.075
.166	.011	-.009	-.003	.005	.012
.277	-.022	-.026	-.025	-.021	-.018
.353			-.036		
.367	-.051	-.051	-.061	-.035	.002
.387	-.022	-.018	-.022	-.024	-.063
.415	-.034				
.443	-.041	-.042	-.040	-.038	-.071
.498	-.045	-.047	-.033	-.038	
.553	-.037	-.043	-.035	-.039	-.054
.581	-.045				
.609	-.050	-.045	-.040	-.036	-.042
.636	-.043				
.664	-.047	-.041	-.035	-.030	-.032
.692	-.048		-.048		
.719	-.044	-.040	-.019	-.037	-.050
.775	-.023	-.017	-.023	-.013	-.014
.830	-.008	-.002	-.015	-.014	-.027
.871		.004	-.006	-.004	.010
.954	.048		.049		
$\alpha = 1^\circ$					
.055	.031		.062		.116
.166	-.021	-.031	-.017	.019	.041
.277	-.037	-.045	-.038	-.012	.004
.353			-.051		
.367	-.051	-.062	-.074	-.033	.023
.387	-.026	-.032	-.041	-.013	-.045
.415	-.036				
.443	-.041	-.046	-.053	-.038	-.054
.498	-.039	-.049	-.044	-.029	
.553	-.033	-.044	-.049	-.036	-.049
.581	-.038				
.609	-.038	-.041	-.053	-.038	-.041
.636	-.032				
.664	-.034	-.039	-.049	-.033	-.030
.692	-.035		-.059		
.719	-.025	-.033	-.030	-.042	-.061
.775	-.008	-.011	-.027	-.017	-.020
.830	.004	-.001	-.024	-.022	-.038
.871		.012	-.018	.019	.008
.954	.058		.044		
$\alpha = 8^\circ$					
.055	.002		.007		.173
.166	-.033	-.063	-.061	.024	.086
.277	-.043	-.068	-.081	-.011	.042
.353			-.082		
.367	-.053	-.074	-.119	-.037	.050
.387	-.028	-.002	-.090	-.034	-.025
.415	-.038				
.443	-.040	-.064	-.098	-.047	-.039
.498	-.037	-.059	-.085	-.050	
.553	-.022	-.054	-.088	-.053	-.033
.581	-.032				
.609	-.038	-.051	-.093	-.049	-.023
.636	-.026				
.664	-.031	-.043	-.090	-.054	-.017
.692	-.036		-.084		
.719	-.025	-.034	-.062	-.060	-.055
.775	-.012	-.015	-.056	-.032	-.014
.830	.030	-.006	-.043	-.045	-.037
.871		.008	-.026	.021	.019
.954	.078		.045		

M = 0.90

Basic body alone					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.070		.074		.083	.055
.002	-.005	-.003	.006	.012	.166
-.026	-.031	-.029	-.026	-.022	.277
		-.045			.353
-.061	-.061	-.074	-.042	.003	.367
-.023	-.018	-.025	-.022	-.067	.387
-.035					.415
-.045	-.050	-.050	-.044	-.076	.443
-.047	-.049	-.040	-.041		.498
-.041	-.048	-.043	-.044	-.061	.553
-.050					.581
-.055	-.048	-.048	-.042	-.052	.609
-.047					.636
-.052	-.048	-.042	-.035	-.037	.664
-.054		-.057			.692
-.031	-.047	-.029	-.045	-.062	.719
-.025	-.021	-.028	-.017	-.019	.775
-.014	-.011	-.023	-.021	-.034	.830
	-.001	-.013	-.008	.002	.871
.051		.049			.954
$\alpha = 4^\circ$					
.041		.071		.126	.055
-.019	-.026	-.015	.021	.045	.166
-.039	-.047	-.041	-.014	.002	.277
		-.058			.353
-.056	-.068	-.084	-.037	.028	.367
-.027	-.030	-.041	-.010	-.050	.387
-.037					.415
-.045	-.051	-.058	-.036	-.058	.443
-.044	-.053	-.048	-.031		.498
-.037	-.049	-.057	-.042	-.055	.553
-.042					.581
-.042	-.047	-.059	-.045	-.048	.609
-.036					.636
-.039	-.045	-.055	-.037	-.035	.664
-.039		-.067			.692
-.029	-.039	-.038	-.051	-.067	.719
-.007	-.015	-.031	-.020	-.021	.775
.001	-.007	-.030	-.028	-.043	.830
	.009	-.019	.020	.008	.871
.060		.045			.954
$\alpha = 8^\circ$					
.011		.019		.184	.055
-.034	-.060	-.059	.025	.088	.166
-.047	-.070	-.084	-.015	.039	.277
		-.076			.353
-.080	-.090	-.134	-.047	.054	.367
-.028	-.005	-.093	-.033	-.030	.387
-.040					.415
-.045	-.069	-.106	-.047	-.044	.443
-.040	-.064	-.090	-.052		.498
-.027	-.059	-.097	-.057	-.039	.553
-.039					.581
-.042	-.055	-.100	-.060	-.033	.609
-.030					.636
-.035	-.050	-.097	-.061	-.023	.664
-.039		-.094			.692
-.029	-.041	-.069	-.072	-.067	.719
-.012	-.020	-.062	-.038	-.018	.775
.024	-.012	-.049	-.052	-.045	.830
	.006	-.032	.023	.018	.871
.082		.043			.954

TABLE IV.- Continued  
PRESSURE COEFFICIENTS FOR BASIC BODY ALONE

M = 0.80

Basic body alone					
X T	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	-.027		-.064		.234
.166	-.045	-.098	-.130	.019	.132
.277	-.043	-.094	-.146	-.019	.081
.353			-.157		
.367	-.054	-.088	-.180	-.052	.085
.387	-.021	-.039	-.159	-.038	.014
.415	-.031				
.443	-.035	-.080	-.157	-.072	.000
.498	-.032	-.075	-.133	-.055	
.553	-.023	-.070	-.141	-.063	-.006
.581	-.036				
.609	-.040	-.064	-.142	-.069	-.005
.636	-.028				
.664	-.034	-.058	-.134	-.071	.003
.692	-.045		-.123		
.719	-.030	-.050	-.093	-.078	-.039
.775	-.014	-.027	-.085	-.049	.002
.830	.021	-.007	-.066	-.063	-.024
.871		-.007	-.047	.022	.023
.954	.071		.032		
$\alpha = 20^\circ$					
.055	-.068		-.295		.380
.166	-.070	-.163	-.357	-.021	.255
.277	-.071	-.157	-.348	-.063	.191
.353			-.346		
.367	-.104	-.158	-.366	-.099	.179
.387	-.069	-.083	-.354	-.115	.107
.415	-.085				
.443	-.082	-.155	-.329	-.074	.080
.498	-.075	-.141	-.286	-.128	
.553	-.055	-.130	-.285	-.133	.073
.581	-.072				
.609	-.077	-.127	-.275	-.129	.072
.636	-.061				
.664	-.071	-.112	-.257	-.134	.071
.692	-.083		-.241		
.719	-.068	-.098	-.180	-.139	.022
.775	-.065	-.071	-.148	-.108	.057
.830	-.017	-.044	-.106	-.125	.028
.871		-.036	-.072	-.003	.076
.954	.026		.013		

M = 0.90

Basic body alone					
Row A	Row B	Row C	Row D	Row E	X T
$\alpha = 12^\circ$					
-.019		-.050		.248	.055
-.044	-.095	-.128	.024	.138	.166
-.047	-.097	-.150	-.019	.082	.277
		-.162			.353
-.060	-.093	-.190	-.053	.090	.367
-.025	-.042	-.163	-.033	.012	.387
-.032					.415
-.036	-.085	-.163	-.076	-.001	.443
-.033	-.081	-.135	-.056		.498
-.026	-.077	-.147	-.068	-.012	.553
-.041					.581
-.043	-.070	-.147	-.076	-.010	.609
-.031					.636
-.037	-.063	-.138	-.076	-.001	.664
-.050		-.131			.692
-.031	-.055	-.098	-.086	-.049	.719
-.010	-.031	-.084	-.051	.000	.775
.016	-.013	-.067	-.068	-.030	.830
	-.010	-.047	.033	.023	.871
.080		.034			.954
$\alpha = 20^\circ$					
-.059		-.278		.399	.055
-.070	-.164	-.362	-.011	.261	.166
-.080	-.166	-.360	-.058	.192	.277
		-.361			.353
-.110	-.168	-.377	-.101	.182	.367
-.084	-.087	-.346	-.092	.110	.387
-.076					.415
-.073	-.157	-.315	-.146	.088	.443
-.064	-.146	-.269	-.118		.498
-.052	-.135	-.275	-.128	.063	.553
-.077					.581
-.073	-.123	-.262	-.134	.065	.609
-.054					.636
-.063	-.114	-.237	-.136	.067	.664
-.083		-.223			.692
-.063	-.103	-.161	-.144	.010	.719
-.046	-.073	-.121	-.102	.054	.775
-.019	-.048	-.091	-.121	.022	.830
	-.035	-.057	.023	.072	.871
.040		.022			.954

TABLE IV.- Continued  
PRESSURE COEFFICIENTS FOR BASIC BODY ALONE

M = 0.94

Basic body alone					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.079		.083		.092
.166	.009	.000	.004	.011	.016
.277	-.024	-.029	-.028	-.025	-.021
.353			-.047		
.367	-.063	-.065	-.078	-.042	.008
.387	-.015	-.015	-.021	-.016	-.067
.415	-.030				
.443	-.045	-.045	-.049	-.044	-.077
.498	-.045	-.048	-.039	-.038	
.553	-.040	-.048	-.044	-.044	-.064
.581	-.048				
.609	-.055	-.048	-.049	-.045	-.054
.636	-.047				
.664	-.054				
.692	-.057	-.049	-.044	-.036	-.039
.719	-.048	-.048	-.059	-.047	-.064
.775	-.022	-.019	-.027	-.017	-.020
.830	-.016	-.015	-.024	-.022	-.036
.871		.002	-.011	-.007	.003
.954	.057		.053		
$\alpha = 4^\circ$					
.055	.046		.077		.133
.166	-.017	-.027	-.013	.023	.045
.277	-.044	-.051	-.044	-.018	-.001
.353			-.062		
.367	-.063	-.075	-.094	-.041	.029
.387	-.026	-.031	-.042	-.010	-.055
.415	-.037				
.443	-.051	-.056	-.064	-.040	-.061
.498	-.047	-.056	-.053	-.034	
.553	-.044	-.056	-.062	-.047	-.061
.581	-.047				
.609	-.049	-.054	-.066	-.053	-.056
.636	-.042				
.664	-.045	-.051	-.058	-.042	-.040
.692	-.046		-.074		
.719	-.036	-.045	-.043	-.058	-.077
.775	-.010	-.018	-.032	-.022	-.025
.830	-.004	-.013	-.036	-.032	-.050
.871		.006	-.023	.018	.004
.954	.060		.046		
$\alpha = 8^\circ$					
.055	.014		.024		.189
.166	-.031	-.059	-.059	.026	.089
.277	-.049	-.075	-.087	-.016	.037
.353			-.078		
.367	-.064	-.086	-.095	-.052	.054
.387	-.026	-.008	-.048	-.030	-.036
.415	-.041				
.443	-.049	-.075	-.113	-.048	-.048
.498	-.041	-.070	-.094	-.052	
.553	-.030	-.064	-.101	-.061	-.047
.581	-.040				
.609	-.045	-.062	-.105	-.066	-.042
.636	-.031				
.664	-.037	-.058	-.101	-.065	-.028
.692	-.045		-.101		
.719	-.031	-.049	-.074	-.080	-.076
.775	-.011	-.025	-.065	-.040	-.023
.830	.019	-.017	-.055	-.057	-.051
.871		.003	-.035	.027	.016
.954	.082		.044		

M = 0.98

Basic body alone					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.089		.091		.099	.055
.009	.002	.002		.015	.166
-.033	-.036	-.036	-.035	-.031	.277
		-.063			.353
-.096	-.098	-.110	-.059	.003	.367
-.010	-.009	-.017	-.015	-.071	.387
-.028					.415
-.053	-.054	-.059	-.052	-.084	.443
-.053	-.056	-.047	-.046		.498
-.051	-.059	-.055	-.056	-.076	.553
-.058					.581
-.066	-.060	-.063	-.061	-.072	.609
-.056					.636
-.063	-.059	-.056	-.047	-.051	.664
-.068		-.075			.692
-.063	-.065	-.049	-.066	-.084	.719
-.026	-.024	-.034	-.022	-.026	.775
-.028	-.029	-.038	-.031	-.047	.830
	-.004	-.018	-.012	-.003	.871
.059		.054			.954
$\alpha = 4^\circ$					
.058		.088		.143	.055
-.015	-.023	-.010	.025	.046	.166
-.049	-.056	-.050	-.023	-.009	.277
		-.071			.353
-.082	-.101	-.120	-.051	.030	.367
-.024	-.023	-.036	-.012	-.064	.387
-.035					.415
-.058	-.061	-.069	-.044	-.061	.443
-.053	-.063	-.062	-.048		.498
-.051	-.063	-.071	-.057	-.072	.553
-.054					.581
-.058	-.062	-.079	-.067	-.070	.609
-.050					.636
-.053	-.058	-.067	-.051	-.048	.664
-.057		-.087			.692
-.047	-.059	-.061	-.079	-.101	.719
-.014	-.021	-.035	-.025	-.027	.775
-.011	-.021	-.048	-.039	-.058	.830
	.004	-.027	.019	.004	.871
.066		.048			.954
$\alpha = 8^\circ$					
.026		.035		.200	.055
-.031	-.058	-.056	.028	.091	.166
-.058	-.085	-.095	-.021	.033	.277
		-.072			.353
-.089	-.119	-.175	-.054	.056	.367
-.020	-.005	-.091	-.040	-.063	.387
-.039					.415
-.062	-.087	-.121	-.046	-.048	.443
-.047	-.073	-.098	-.070		.498
-.040	-.076	-.114	-.070	-.057	.553
-.052					.581
-.056	-.074	-.123	-.078	-.053	.609
-.041					.636
-.048	-.070	-.109	-.074	-.037	.664
-.059		-.115			.692
-.047	-.065	-.094	-.104	-.102	.719
-.015	-.030	-.066	-.045	-.030	.775
.009	-.028	-.068	-.066	-.057	.830
	-.003	-.040	.036	.018	.871
.084		.043			.954

TABLE IV.- Continued  
PRESSURE COEFFICIENTS FOR BASIC BODY ALONE

$M = 0.94$

Basic body alone					
$\frac{x}{T}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	-.013		-.043		.254
.166	-.044	-.096	-.128	.024	.140
.277	-.051	-.102	-.154	-.022	.079
.353			-.168		
.367	-.066	-.100	-.210	-.058	.088
.387	-.028	-.045	-.168	-.036	.003
.415	-.035				
.443	-.042	-.093	-.169	-.075	-.005
.498	-.037	-.088	-.137	-.060	
.553	-.032	-.084	-.155	-.075	-.020
.581	-.048				
.609	-.049	-.077	-.156	-.085	-.020
.636	-.036				
.664	-.043	-.071	-.144	-.083	-.009
.692	-.057		-.142		
.719	-.037	-.062	-.105	-.097	-.061
.775	-.014	-.035	-.085	-.055	-.005
.830	.010	-.020	-.074	-.075	-.038
.871		-.016	-.054	.039	.021
.954	.080		.032		
$\alpha = 20^\circ$					
.055	-.050		-.266		.407
.166	-.069	-.162	-.352	-.006	.267
.277	-.081	-.171	-.368	-.055	.194
.353			-.365		
.367	-.116	-.177	-.430	-.098	.180
.387	-.118	-.089	-.352	-.091	.104
.415	-.078				
.443	-.071	-.167	-.329	-.164	.087
.498	-.065	-.153	-.273	-.123	
.553	-.062	-.142	-.282	-.136	.058
.581	-.088				
.609	-.078	-.130	-.283	-.137	.058
.636	-.066				
.664	-.071	-.122	-.237	-.143	.061
.692	-.094		-.240		
.719	-.068	-.108	-.163	-.153	-.002
.775	-.047	-.077	-.118	-.103	.050
.830	-.021	-.055	-.093	-.125	.016
.871		-.037	-.061	.040	.072
.954	.045		.025		

$M = 0.98$

Basic body alone					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{T}$
$\alpha = 12^\circ$					
.000		-.028		.266	.055
-.041	-.091	-.124	.029	.143	.166
-.057	-.109	-.155	-.025	.078	.277
		-.177			.353
-.094	-.129	-.230	-.054	.092	.367
-.029	-.046	-.183	-.048	-.018	.387
-.027					.415
-.050	-.105	-.164	-.071	.003	.443
-.042	-.100	-.155	-.088		.498
-.041	-.096	-.159	-.070	-.021	.553
-.061					.581
-.058	-.088	-.177	-.099	-.036	.609
-.039					.636
-.051	-.079	-.142	-.089	-.013	.664
-.077		-.158			.692
-.054	-.078	-.128	-.127	-.090	.719
-.014	-.038	-.081	-.058	-.014	.775
.003	-.029	-.084	-.078	-.040	.830
	-.022	-.057	.057	.024	.871
.090		.037			.954
$\alpha = 20^\circ$					
-.036		-.245		.425	.055
-.062	-.155	-.359	.007	.277	.166
-.081	-.170	-.371	-.054	.198	.277
		-.379			.353
-.151	-.213	-.430	-.090	.192	.367
-.122	-.105	-.393	-.104	.097	.387
-.051					.415
-.081	-.155	-.284	-.164	.084	.443
-.104	-.178	-.308	-.119		.498
-.082	-.152	-.300	-.166	.044	.553
-.100					.581
-.072	-.138	-.279	-.175	.043	.609
-.068					.636
-.095	-.134	-.241	-.137	.063	.664
-.124		-.246			.692
-.071	-.129	-.196	-.167	-.008	.719
-.040	-.079	-.108	-.136	.018	.775
-.027	-.068	-.097	-.124	.021	.830
	-.047	-.065	.068	.068	.871
.058		.033			.954

TABLE IV.- Continued  
PRESSURE COEFFICIENTS FOR BASIC BODY ALONE

M = 1.03

Basic body alone					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 0^\circ$					
.055	.119		.122		.130
.166	.054	.048	.048	.056	.060
.277	.013	.011	.011	.012	.016
.353			.032		
.367	-.069	-.073	.091	-.048	.013
.387	-.037	-.029	-.023	-.012	-.074
.415	-.023				
.443	-.048	-.048	-.056	-.046	-.076
.498	-.077	-.079	-.070	-.071	
.553	-.079	-.088	-.083	-.081	-.101
.581	-.087				
.609	-.096	-.088	-.094	-.089	-.100
.636	-.091				
.664	-.098	-.095	-.091	-.088	-.092
.692	-.108		-.112		
.719	-.108	-.109	-.090	-.110	-.127
.775	-.087	-.084	-.092	-.086	-.090
.830	-.078	-.079	-.085	-.076	-.093
.871		-.075	-.084	-.073	-.073
.954	.061		.057		
$\alpha = 4^\circ$					
.055	.092		.124		.178
.166	.033	.026	.038	.072	.094
.277	-.002	-.008	.000	.027	.044
.353			-.041		
.367	-.069	-.086	-.094	-.025	.055
.387	-.042	-.038	-.039	.000	-.043
.415	-.030				
.443	-.051	-.053	-.063	-.033	-.052
.498	-.073	-.083	-.079	-.064	
.553	-.078	-.091	-.095	-.079	-.094
.581	-.083				
.609	-.087	-.091	-.106	-.090	-.096
.636	-.079				
.664	-.087	-.093	-.100	-.093	-.092
.692	-.095		-.120		
.719	-.091	-.102	-.099	-.112	-.133
.775	-.069	-.079	-.097	-.094	-.102
.830	-.063	-.075	-.094	-.077	-.096
.871		-.057	-.083	-.024	-.061
.954	.058		.038		
$\alpha = 8^\circ$					
.055	.062		.072		.212
.166	.020	-.006	-.008	.073	.134
.277	-.012	-.036	-.046	.024	.077
.353			-.026		
.367	-.081	-.108	-.140	-.024	.083
.387	-.034	-.017	-.098	-.018	-.028
.415	-.038				
.443	-.058	-.086	-.117	-.041	-.042
.498	-.072	-.102	-.128	-.084	
.553	-.068	-.104	-.141	-.095	-.077
.581	-.080				
.609	-.085	-.103	-.153	-.107	-.086
.636	-.074				
.664	-.084	-.106	-.147	-.122	-.089
.692	-.101		-.152		
.719	-.093	-.111	-.136	-.138	-.134
.775	-.071	-.090	-.130	-.121	-.110
.830	-.044	-.088	-.119	-.104	-.101
.871		-.062	-.089	-.005	-.058
.954	.101		.047		

M = 1.115

Basic body alone					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 0^\circ$					
.096		.100		.106	.055
.027	.022	.020	.025	.029	.166
-.001	-.004	-.001	.000	.003	.277
		-.027			.353
-.045	-.053	-.067	-.032	.014	.367
-.027	-.028	-.022	.001	-.040	.387
-.023					.415
-.019	-.020	-.037	-.031	-.052	.443
-.024	-.034	-.033	-.040		.498
-.033	-.039	-.039	-.042	-.061	.553
-.046					.581
-.054	-.047	-.054	-.051	-.063	.609
-.060					.636
-.069	-.065	-.060	-.058	-.060	.664
-.075		-.082			.692
-.059	-.070	-.057	-.074	-.091	.719
-.032	-.035	-.046	-.039	-.044	.775
-.050	-.051	-.055	-.040	-.058	.830
	-.049	-.054	-.031	-.041	.871
-.037		-.045			.954
$\alpha = 4^\circ$					
.069		.099		.152	.055
.006	-.001	.009	.045	.067	.166
-.017	-.024	-.015	.015	.032	.277
		-.035			.353
-.056	-.074	-.079	-.020	.033	.367
-.030	-.042	-.045	.005	-.006	.387
-.020					.415
-.029	-.035	-.057	-.026	-.036	.443
-.036	-.045	-.041	-.022		.498
-.034	-.048	-.052	-.036	-.047	.553
-.044					.581
-.050	-.055	-.067	-.051	-.060	.609
-.051					.636
-.057	-.066	-.075	-.069	-.072	.664
-.061		-.088			.692
-.051	-.063	-.059	-.072	-.098	.719
-.027	-.039	-.054	-.045	-.063	.775
-.041	-.053	-.068	-.049	-.066	.830
	-.036	-.056	-.009	-.048	.871
-.017		-.041			.954
$\alpha = 8^\circ$					
.044		.053		.215	.055
-.004	-.027	-.019	.058	.120	.166
-.020	-.042	-.051	.020	.072	.277
		.114			.353
-.060	-.085	-.116	-.028	.049	.367
-.001	-.022	-.086	-.004	.015	.387
-.014					.415
.005	-.039	-.079	-.009	-.004	.443
-.016	-.048	-.081	-.039		.498
-.026	-.062	-.093	-.050	-.038	.553
-.047					.581
-.045	-.066	-.108	-.068	-.046	.609
-.035					.636
-.030	-.062	-.106	-.089	-.055	.664
-.039		-.107			.692
-.025	-.055	-.072	-.082	-.085	.719
-.032	-.046	-.082	-.072	-.059	.775
.000	-.048	-.078	-.064	-.062	.830
	-.039	-.055	.018	-.029	.871
.033		-.044			.954

TABLE IV.- Concluded  
PRESSURE COEFFICIENTS FOR BASIC BODY ALONE

M = 1.03

Basic body alone					
$\frac{x}{l}$	Row A	Row B	Row C	Row D	Row E
$\alpha = 12^\circ$					
.055	.036		.007		.300
.166	.004	-.046	-.082	.069	.182
.277	-.017	-.068	-.117	.019	.120
.353			-.154		
.367	-.082	-.120	-.194	-.018	.129
.387	-.042	-.056	-.181	-.022	.022
.415	-.033				
.443	-.044	-.102	-.159	-.069	-.001
.498	-.064	-.122	-.174	-.101	
.553	-.078	-.126	-.196	-.117	-.053
.581	-.100				
.609	-.093	-.125	-.210	-.133	-.063
.636	-.092				
.664	-.105	-.124	-.196	-.148	-.068
.692	-.126		-.201		
.719	-.101	-.130	-.177	-.157	-.118
.775	-.091	-.107	-.159	-.151	-.100
.830	-.072	-.094	-.143	-.131	-.097
.871		-.091	-.107	.002	-.057
.954	.079		.001		
$\alpha = 20^\circ$					
.055	.014		-.193		.465
.166	-.020		-.305	.048	.315
.277	-.051	-.110	-.334	-.011	.237
.353		-.144	-.359		
.367	-.127	-.198	-.400	-.056	.228
.387	-.111	-.109	-.400	-.092	.123
.415	-.095				
.443	-.089	-.169	-.315	-.125	.093
.498	-.093	-.183	-.312	-.140	
.553	-.120	-.181	-.318	-.177	.035
.581	-.144				
.609	-.126	-.174	-.330	-.191	.034
.636	-.124				
.664	-.136	-.172	-.290	-.195	.010
.692	-.168		-.294		
.719	-.135	-.181	-.248	-.208	-.057
.775	-.125	-.152	-.195	-.226	-.050
.830	-.092	-.140	-.165	-.203	-.048
.871		-.112	-.120	-.045	.005
.954	.059		.001		

M = 1.115

Basic body alone					
Row A	Row B	Row C	Row D	Row E	$\frac{x}{l}$
$\alpha = 12^\circ$					
.011		-.012		.279	.055
-.024	-.072	-.103		.168	.166
-.026	-.077	-.122	.055	.110	.277
		-.156	.008		.353
-.067	-.114	-.190	-.034	.089	.367
-.015	-.054	-.176	-.026	.055	.387
-.014					.415
-.004	-.073	-.132	-.036	.012	.443
-.025	-.082	-.148	-.056		.498
-.046	-.088	-.160	-.078	-.011	.553
-.065					.581
-.056	-.090	-.175	-.109	-.032	.609
-.051					.636
-.057	-.077	-.151	-.116	-.045	.664
-.070		-.155			.692
-.045	-.077	-.124	-.105	-.070	.719
-.065	-.068	-.128	-.104	-.043	.775
-.031	-.058	-.103	-.102	-.067	.830
	-.050	-.073	.003	-.023	.871
-.054		-.073			.954
$\alpha = 20^\circ$					
-.012		-.206		.452	.055
-.041	-.130	-.320	.056	.314	.166
-.056	-.138	-.340	-.009	.240	.277
		-.333			.353
-.106	-.175	-.359	-.037	.238	.367
-.047	-.088	-.369	-.075	.159	.387
-.043					.415
-.066	-.128	-.294	-.003	.138	.443
-.075	-.144	-.295	-.127		.498
-.087	-.140	-.283	-.143	.075	.553
-.119					.581
-.087	-.123	-.266	-.152	.048	.609
-.089					.636
-.094	-.125	-.228	-.133	.053	.664
-.116		-.248			.692
-.108	-.131	-.212	-.148	.006	.719
-.085	-.106	-.156	-.199	-.004	.775
-.049	-.107	-.116	-.163	-.006	.830
	-.077	-.077	-.066	.048	.871
-.119		-.070			.954

TABLE V

## TABULATED WING SECTION DATA

M	$\alpha$ , deg	Basic				Elliptical			
		Section normal-force coefficient, $C_n$		Section pitching-moment coefficient, $C_{m,c/4}$		Section normal-force coefficient, $C_n$		Section pitching-moment coefficient, $C_{m,c/4}$	
		$0.20 \frac{b}{L}$	$0.40 \frac{b}{L}$	$0.70 \frac{b}{L}$	$0.20 \frac{b}{L}$	$0.40 \frac{b}{L}$	$0.70 \frac{b}{L}$	$0.20 \frac{b}{L}$	$0.40 \frac{b}{L}$
0.80	0	0.0013	0.0213	0.0116	0.0016	-0.0018	-0.0038	-0.0033	-0.0038
	4	.2013	.2819	.3271	-.0090	-.0087	.0148	-.0111	.0010
	8	.4299	.5890	.6307	-.0184	-.0765	.6271	-.0134	.0003
	12	.6942	.8419	.7729	-.0315	-.1060	.7632	-.0298	-.0665
.90	20	.8787	.8400	.8342	-.1417	-.1368	.8497	-.1481	-.1014
	0	.0045	.0168	.0039	-.0011	-.0075	.0245	-.0049	-.1358
	4	.2232	.2961	.3594	-.0102	-.0170	.3581	-.0121	-.0052
	8	.5090	.6516	.7768	-.0337	-.0670	.7852	-.0306	-.0136
.94	12	.7348	.9168	.9270	-.0657	-.1355	.9632	-.0664	-.0654
	20	1.0084	.9374	.9826	-.1750	-.1837	1.0045	-.1839	-.1150
	0	.0071	.0206	.0058	.0002	-.0062	.0252	-.0041	-.1767
	4	.2484	.3032	.3910	-.0228	-.0008	.4077	-.0121	-.0039
.98	8	.5652	.7394	.8858	-.0685	-.0798	.8658	-.0772	-.0174
	12	.8135	.9800	1.0658	-.1067	-.1504	1.2219	-.1177	-.0751
	20	1.1568	1.2097	1.1703	-.2052	-.2273	1.1794	-.2086	-.1439
	0	.0013	.0116	-.0065	.0003	-.0016	.0206	-.0043	-.2365
1.03	4	.2665	.3490	.4155	-.0401	-.0095	.4077	-.0439	-.0077
	8	.5826	.7316	.8729	-.0982	-.0926	.8561	-.1011	-.0375
	12	.8342	1.0200	1.1452	-.1396	-.1609	1.1832	-.1336	-.0964
	20	1.2226	1.4929	1.4335	-.2019	-.2861	1.4264	-.1971	-.1663
1.03	0	0	.0194	-.0019	.0007	-.0025	.0226	-.0051	-.2589
	4	.2723	.3323	.4284	-.0439	-.0056	.4084	-.0531	-.0064
	8	.5768	.6794	.8458	-.1005	-.0985	.8181	-.0991	-.0446
	12	.8019	.9794	1.1426	-.1342	-.1675	1.1258	-.1326	-.0914
1.115	20	1.1903	1.4535	1.4077	-.1975	-.2802	1.4110	-.1899	-.1717
	0	.0090	.0174	.0013	.0028	0	.0109	-.0013	-.2412
	4	.2690	.3226	.3581	-.0519	-.0341	.3716	-.0582	-.0049
	8	.5419	.6452	.7839	-.0977	-.1009	.7742	-.0983	-.0534
1.115	12	.7735	.9084	1.0968	-.1332	-.1760	1.0748	-.1314	-.0926
	16.6	1.0213	1.2381	1.3084	-.2099	-.2399	1.3490	-.1845	-.1437
	20								-.2637



TABLE VI

## WING TWIST INFLUENCE COEFFICIENTS

Twist measurement station, $\frac{y}{b/2}$	Rate of change in twist angle due to normal force at section quarter chord, $\frac{\partial \alpha}{\partial n}$ , deg/lb, at -			
	$\frac{y}{b/2} = 0.20$	$\frac{y}{b/2} = 0.40$	$\frac{y}{b/2} = 0.70$	$\frac{y}{b/2} = 0.90$
0.20	0	0	0.00016	-0.00008
.40	.00006	.00053	-.00046	-.00161
.70	.00008	.00195	.00388	-.00259
.90	.00017	.00200	.00696	.00760
1.00	.00007	.00186	.00710	.01200
Twist measurement station, $\frac{y}{b/2}$	Rate of change in twist angle due to pitching moment about section quarter chord, $\frac{\partial \alpha}{\partial m}$ , deg/in-lb, at -			
	$\frac{y}{b/2} = 0.20$	$\frac{y}{b/2} = 0.40$	$\frac{y}{b/2} = 0.70$	$\frac{y}{b/2} = 0.90$
0.20	0.00004	0.00006	0.00012	0
.40	.00009	.00051	.00073	.00045
.70	.00011	.00131	.00619	.00806
.90	.00013	.00140	.00879	.02664
1.00	.00010	.00135	.00930	.03509

Semispan station, $\frac{y}{b/2}$	Exposed semispan station, measured from mounting block outer face, $\frac{y'}{b'/2}$	Local section chord, c, in.
0.20	0.055	9.580
.40	.291	7.785
.70	.646	5.092
.90	.882	3.297

TABLE VII

## NORMAL-FORCE AND PITCHING-MOMENT CHARACTERISTICS FOR BASIC BODY ALONE

M	$\alpha$ , deg	Body normal-force coefficient, $C_{N,f}$	Body pitching-moment coefficient, $C_{m,f}$
0.80	0	0.0024	-0.0004
	4	.0094	.0127
	8	.0136	.0289
	12	.0254	.0372
	20	.0667	.0464
.90	0	.0006	.0022
	4	.0083	.0153
	8	.0130	.0280
	12	.0266	.0376
	20	.0690	.0495
.94	0	-.0012	-.0035
	4	.0065	.0136
	8	.0136	.0267
	12	.0248	.0394
	20	.0685	.0538
.98	0	-.0024	0
	4	.0065	.0127
	8	.0148	.0298
	12	.0271	.0403
	20	.0702	.0512
1.03	0	-.0006	-.0018
	4	.0094	.0153
	8	.0153	.0258
	12	.0289	.0394
	20	.0755	.0547
1.115	0	-.0018	-.0031
	4	.0106	.0153
	8	.0148	.0324
	12	.0271	.0394
	20	.0797	.0565

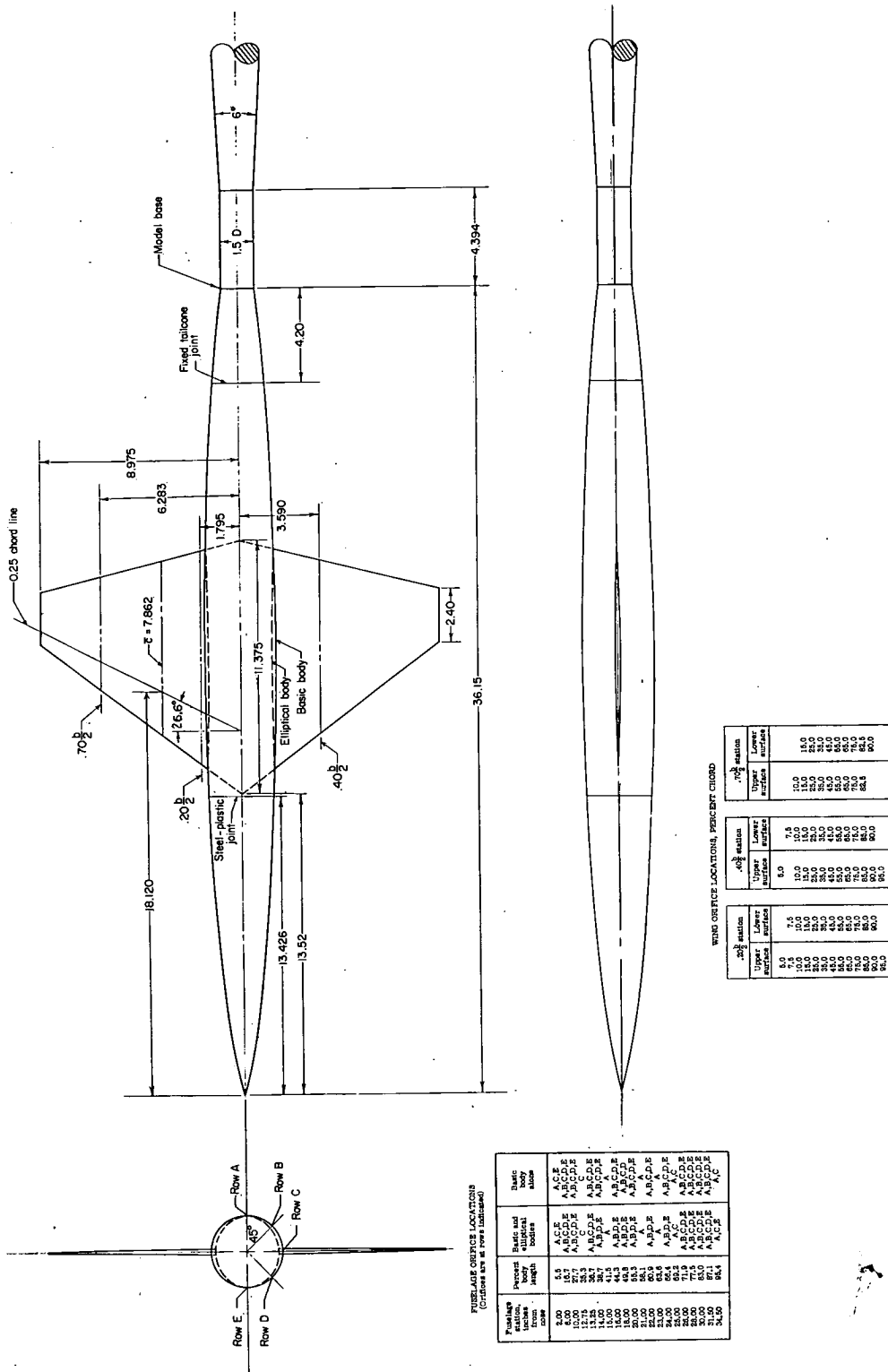


Figure 1.- Model details. All dimensions in inches unless otherwise noted.

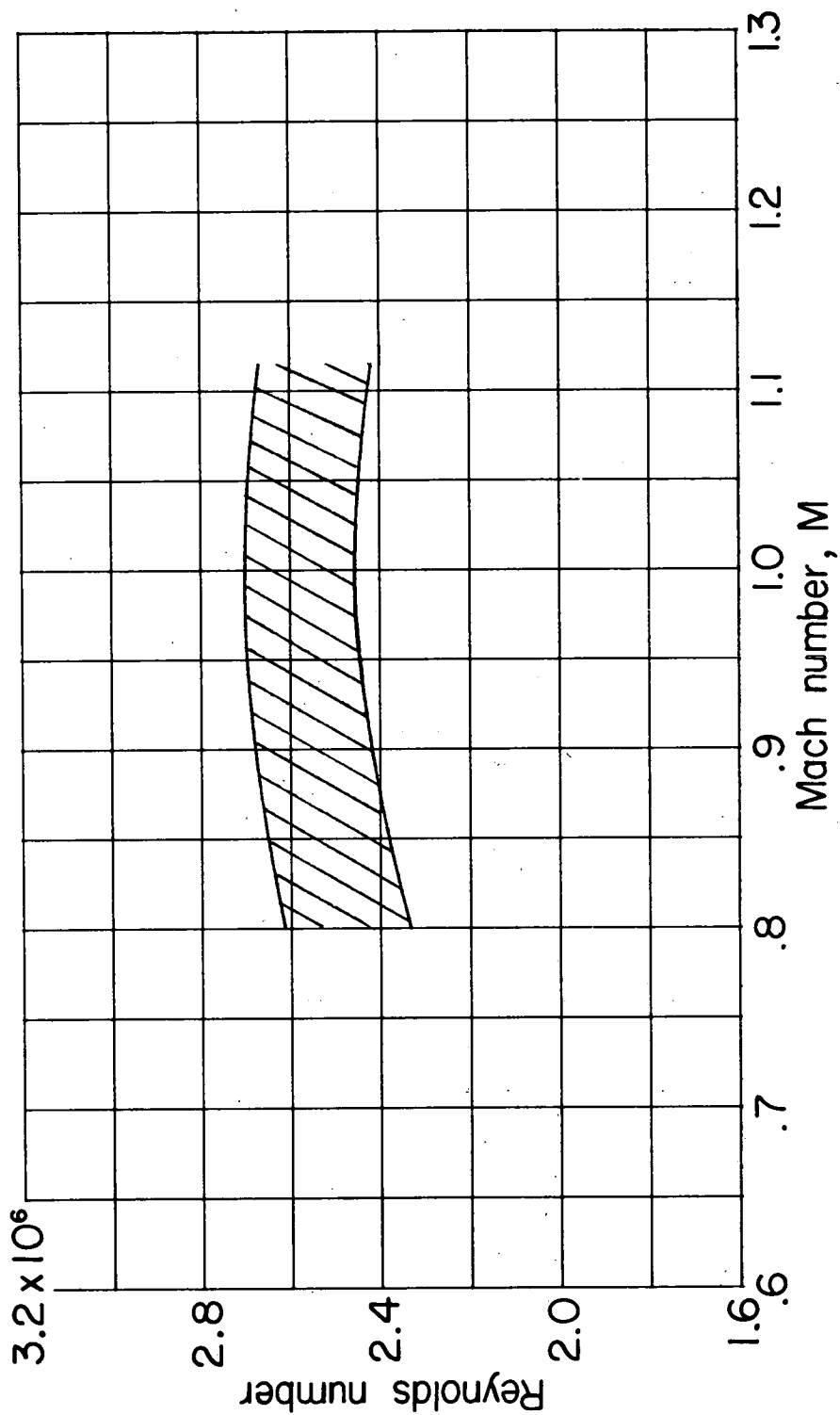


Figure 2.- Variation with Mach number of test Reynolds number, based on  $\bar{c} = 7.86$  inches.

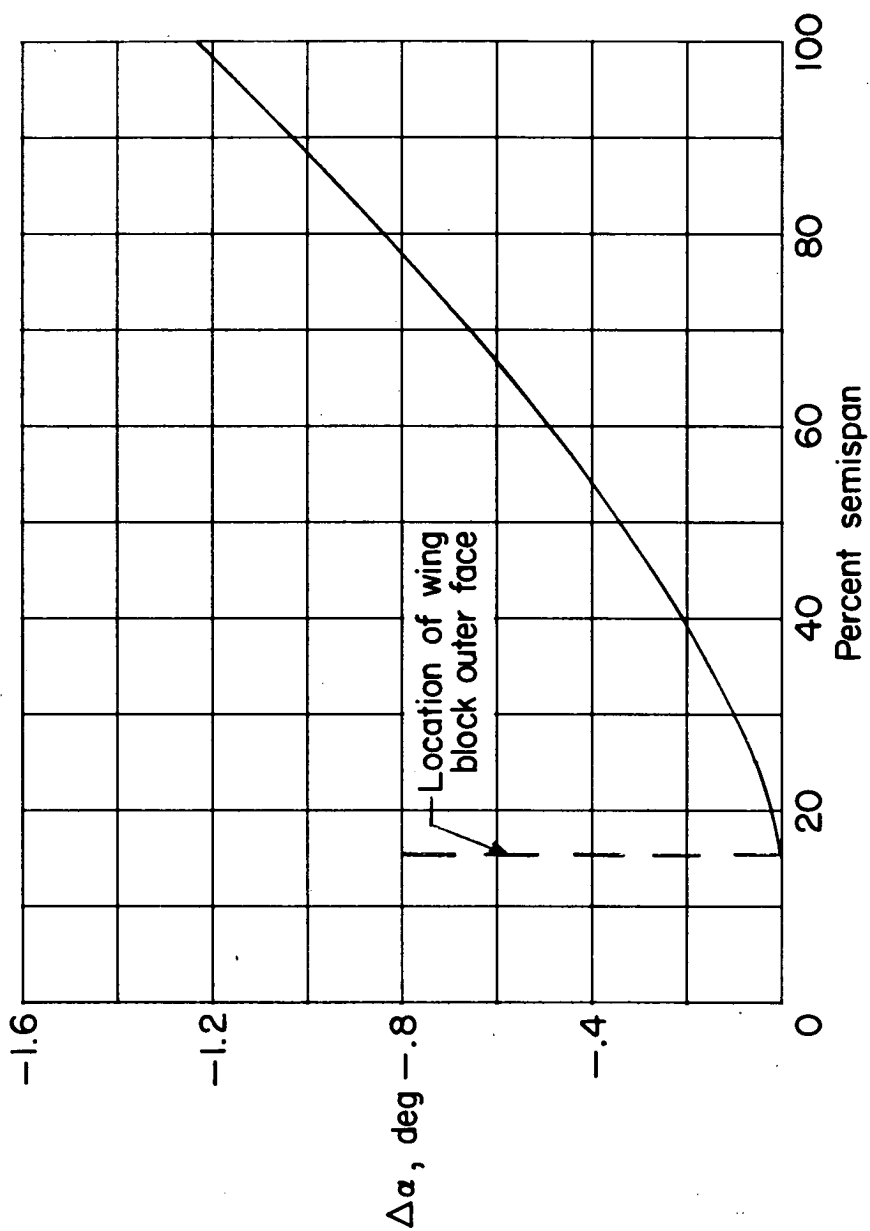


Figure 3.- Spanwise wing twist distribution for elliptical configuration,  
 $M = 1.115$ ,  $\alpha = 20^\circ$ .

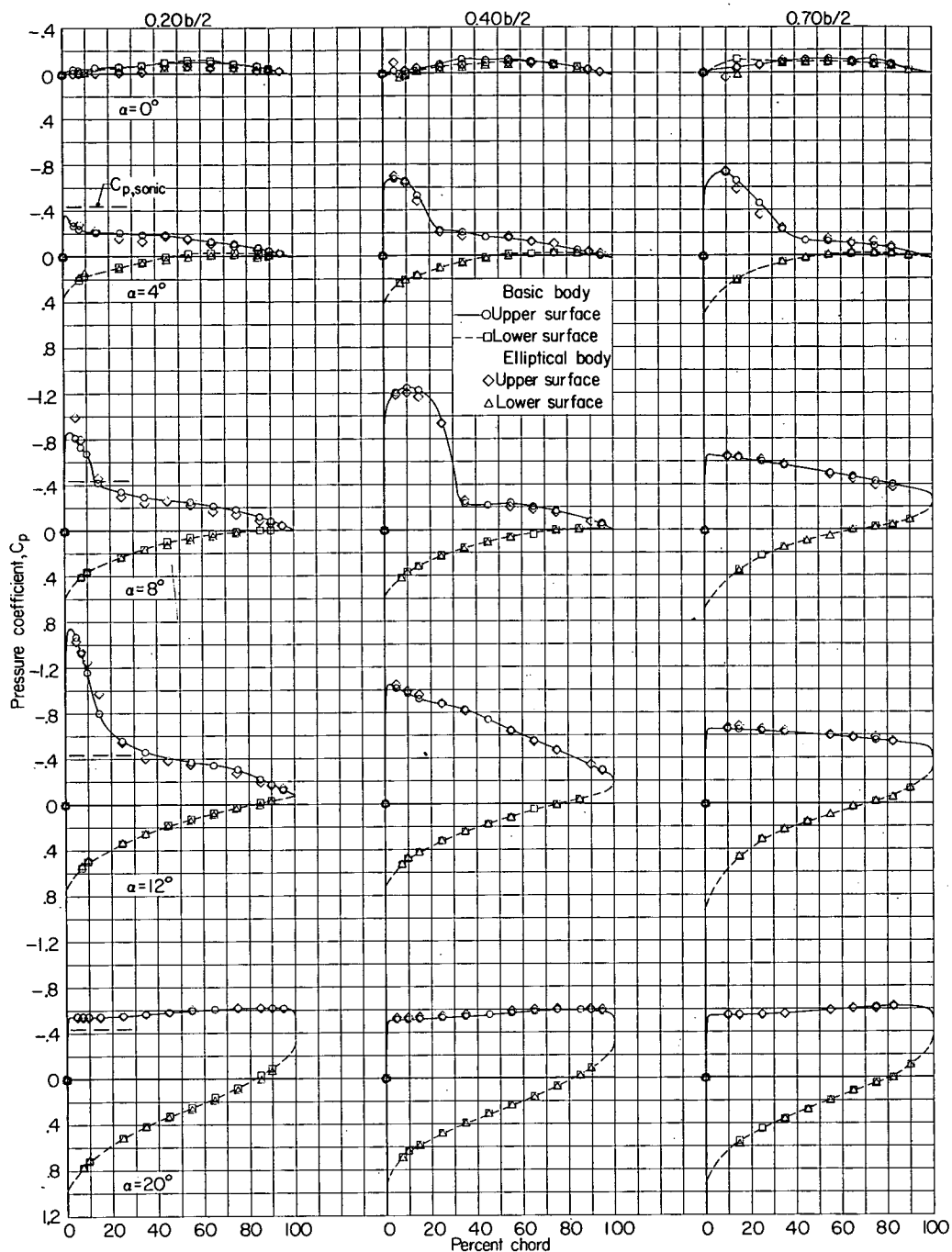
(a)  $M = 0.80$ .

Figure 4.- Pressure coefficients for wing in presence of bodies.

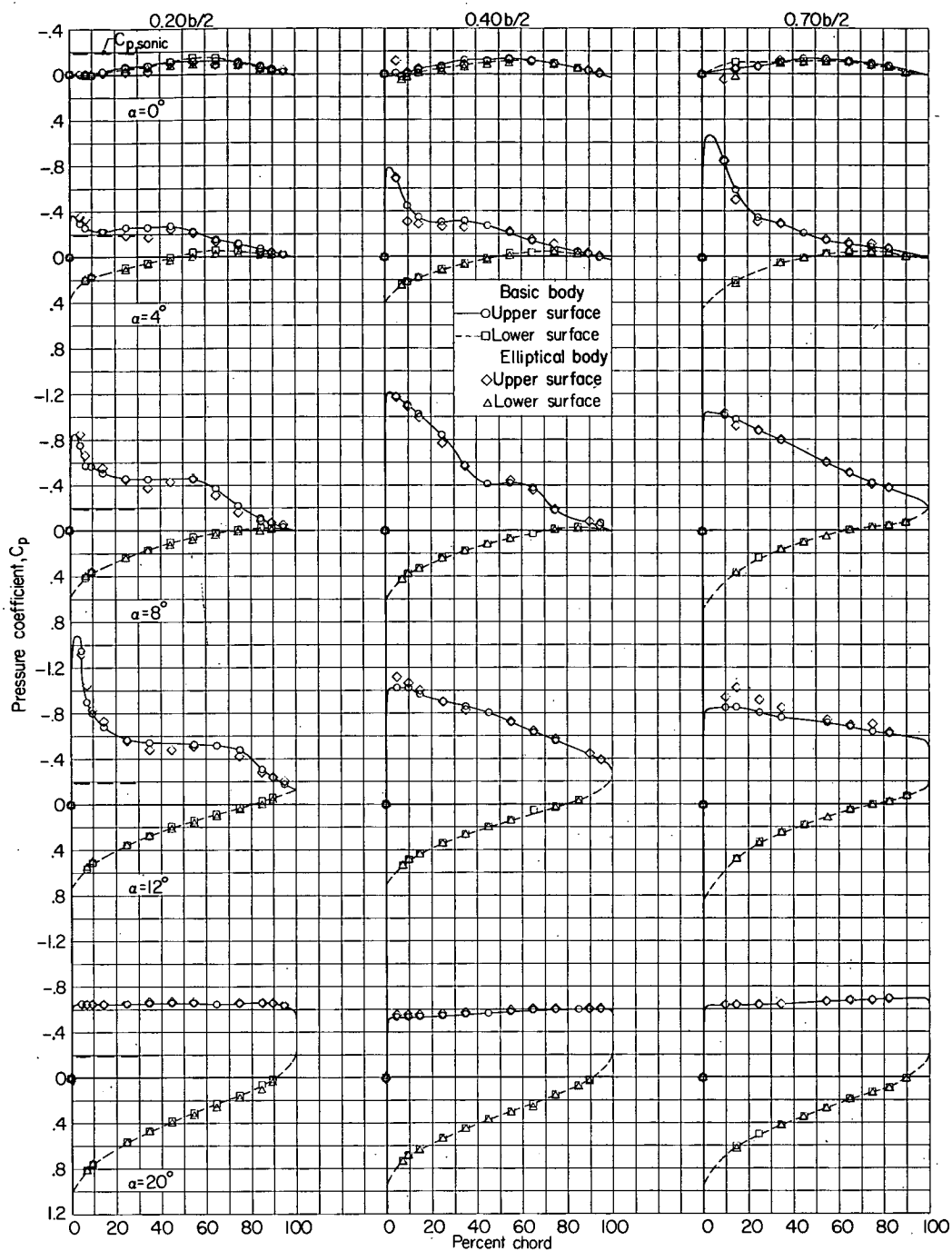
(b)  $M = 0.90$ .

Figure 4.- Continued.

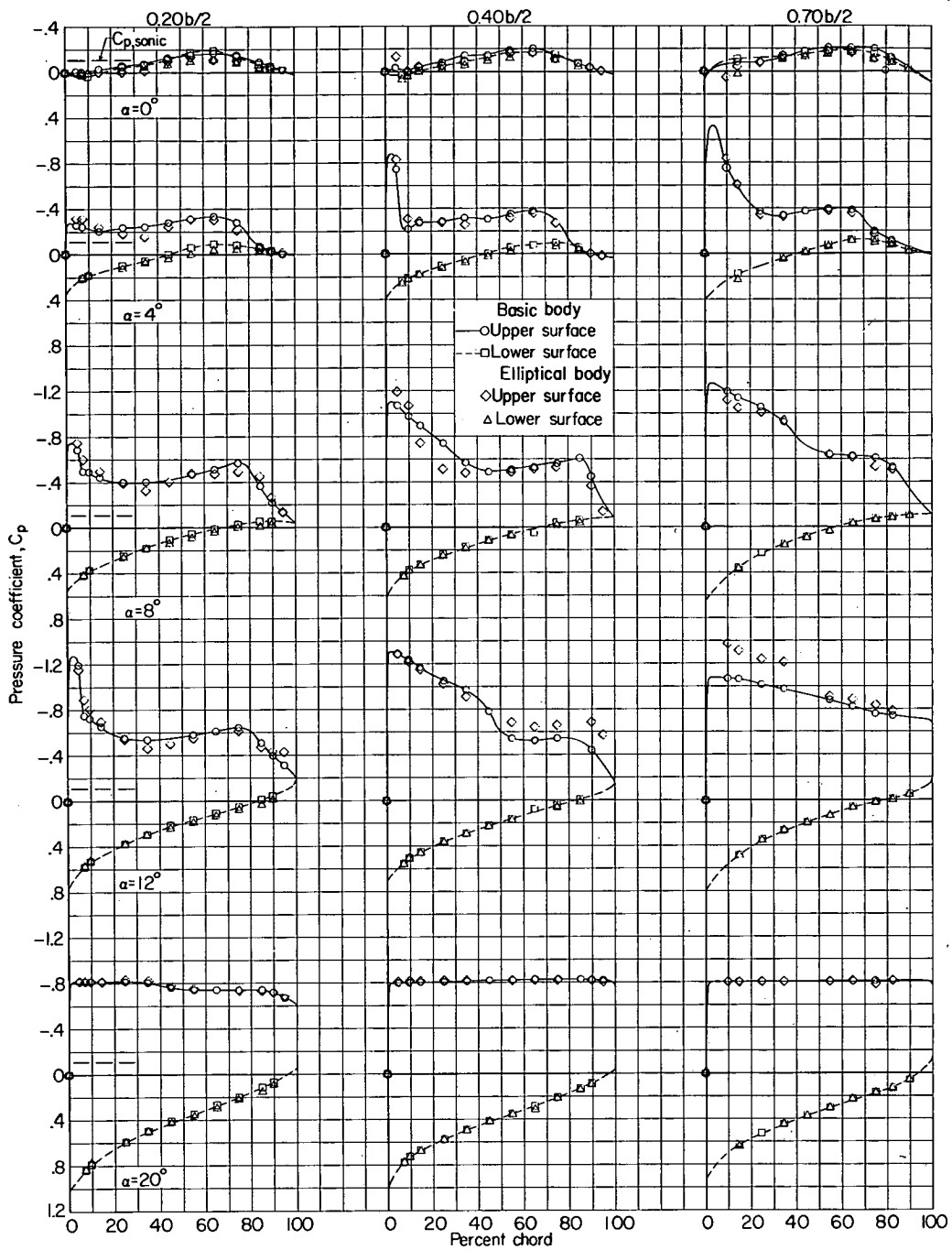
(c)  $M = 0.94$ .

Figure 4.- Continued.



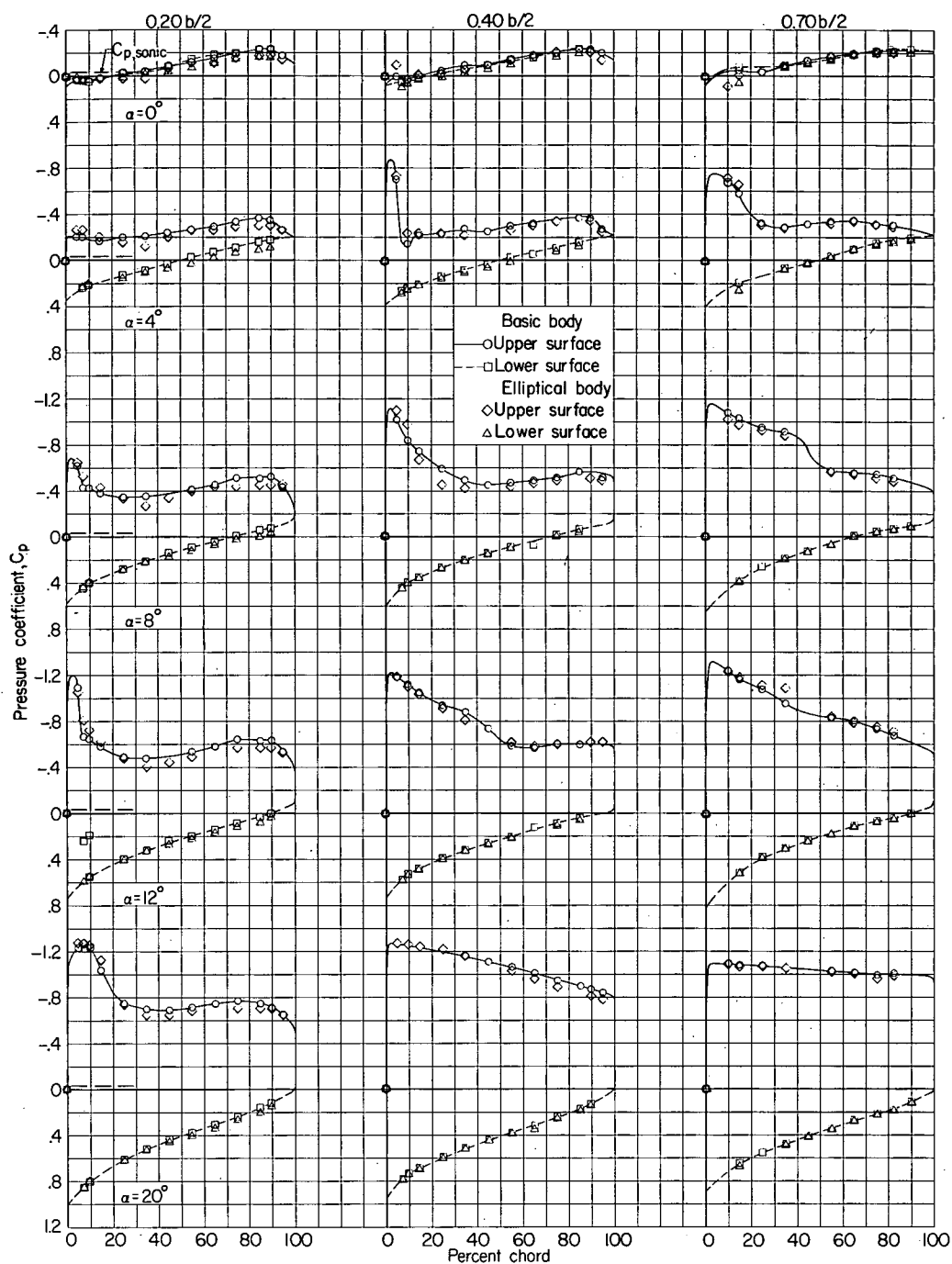
(d)  $M = 0.98$ .

Figure 4.- Continued.

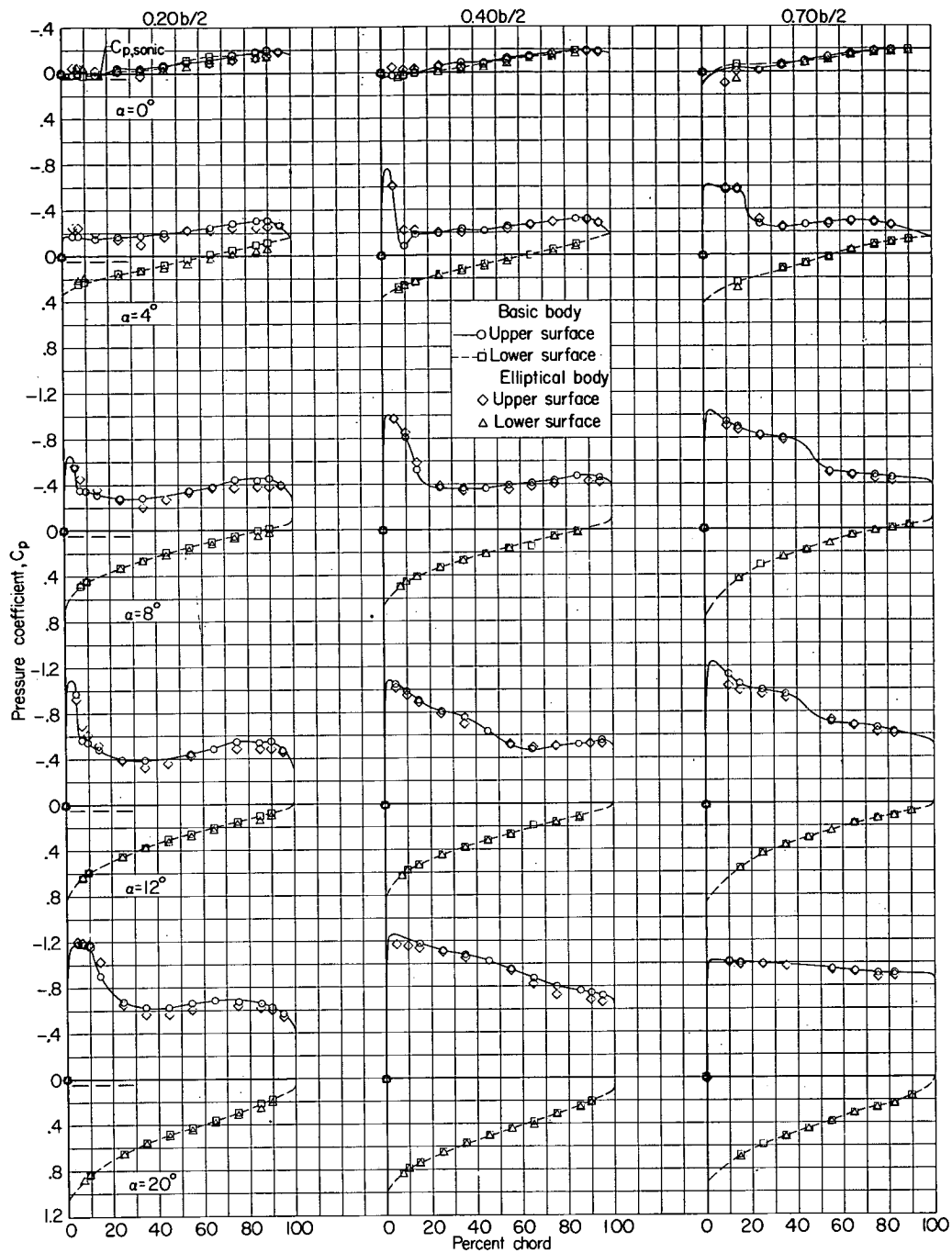
(e)  $M = 1.03$ .

Figure 4.- Continued.

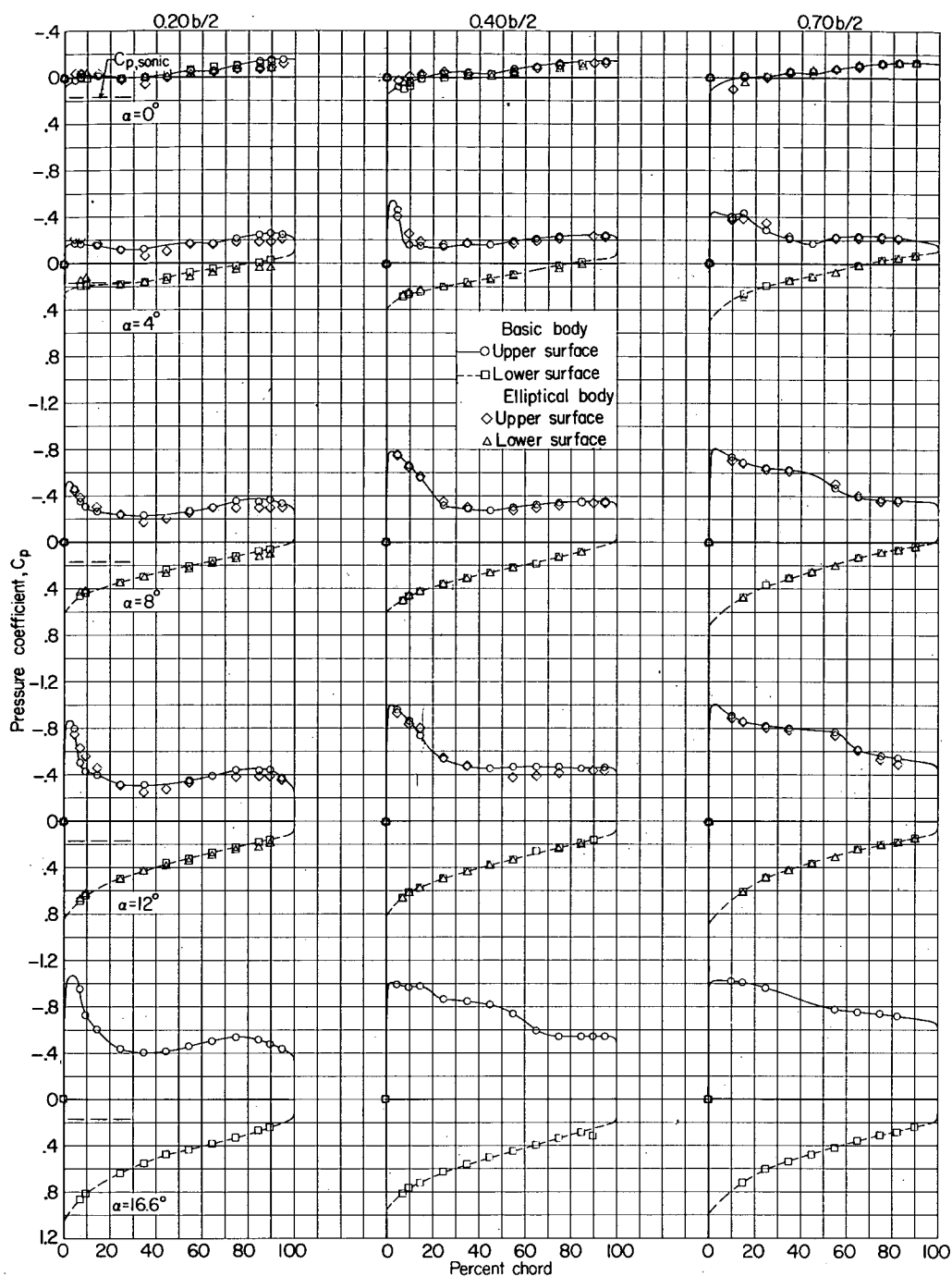
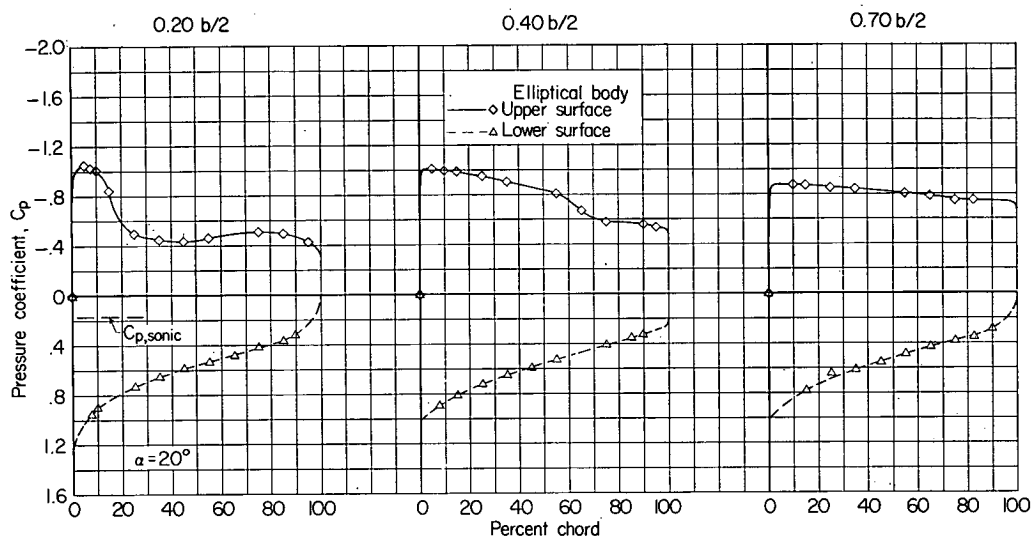
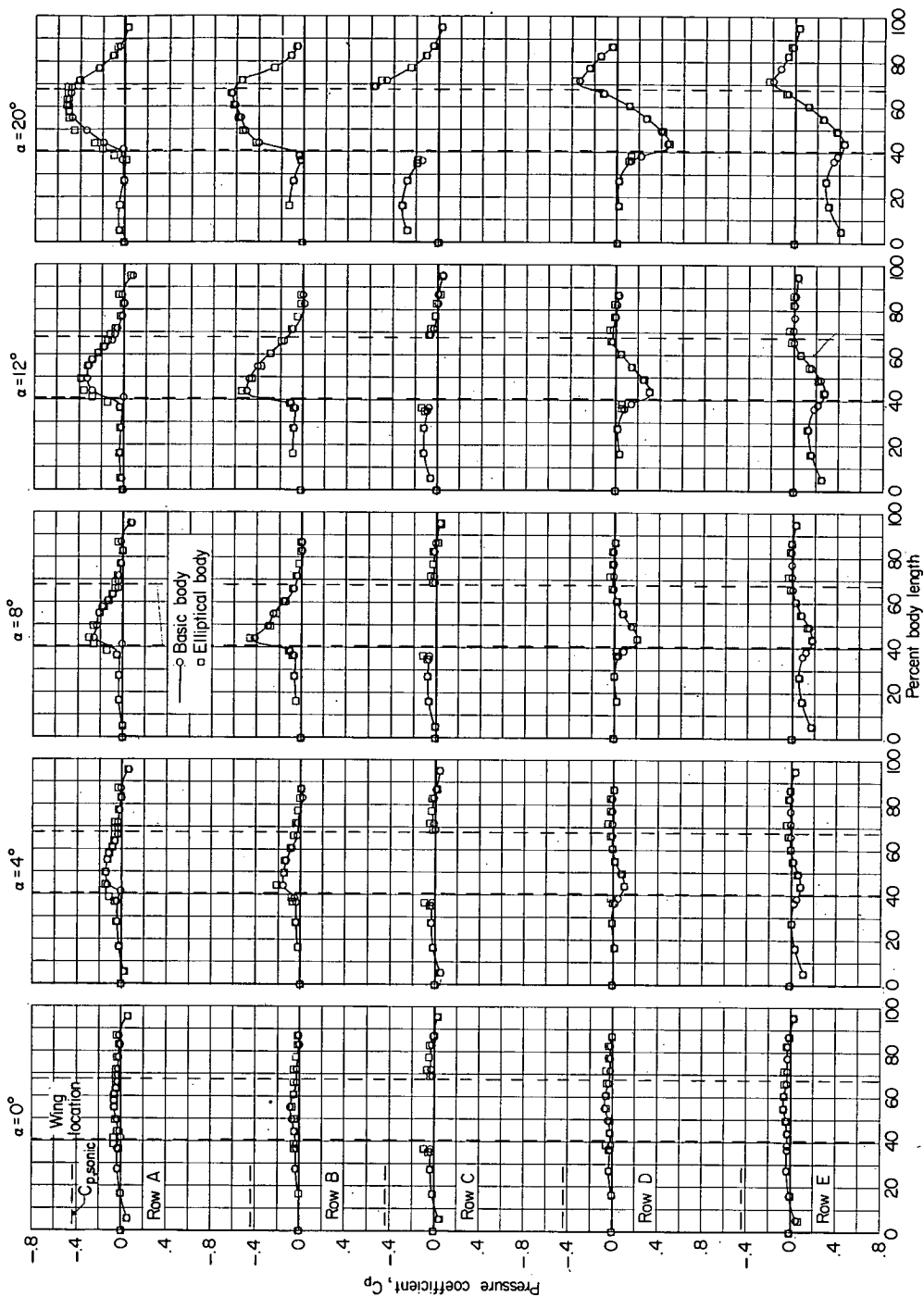
(f)  $M = 1.115$ .

Figure 4.- Continued.



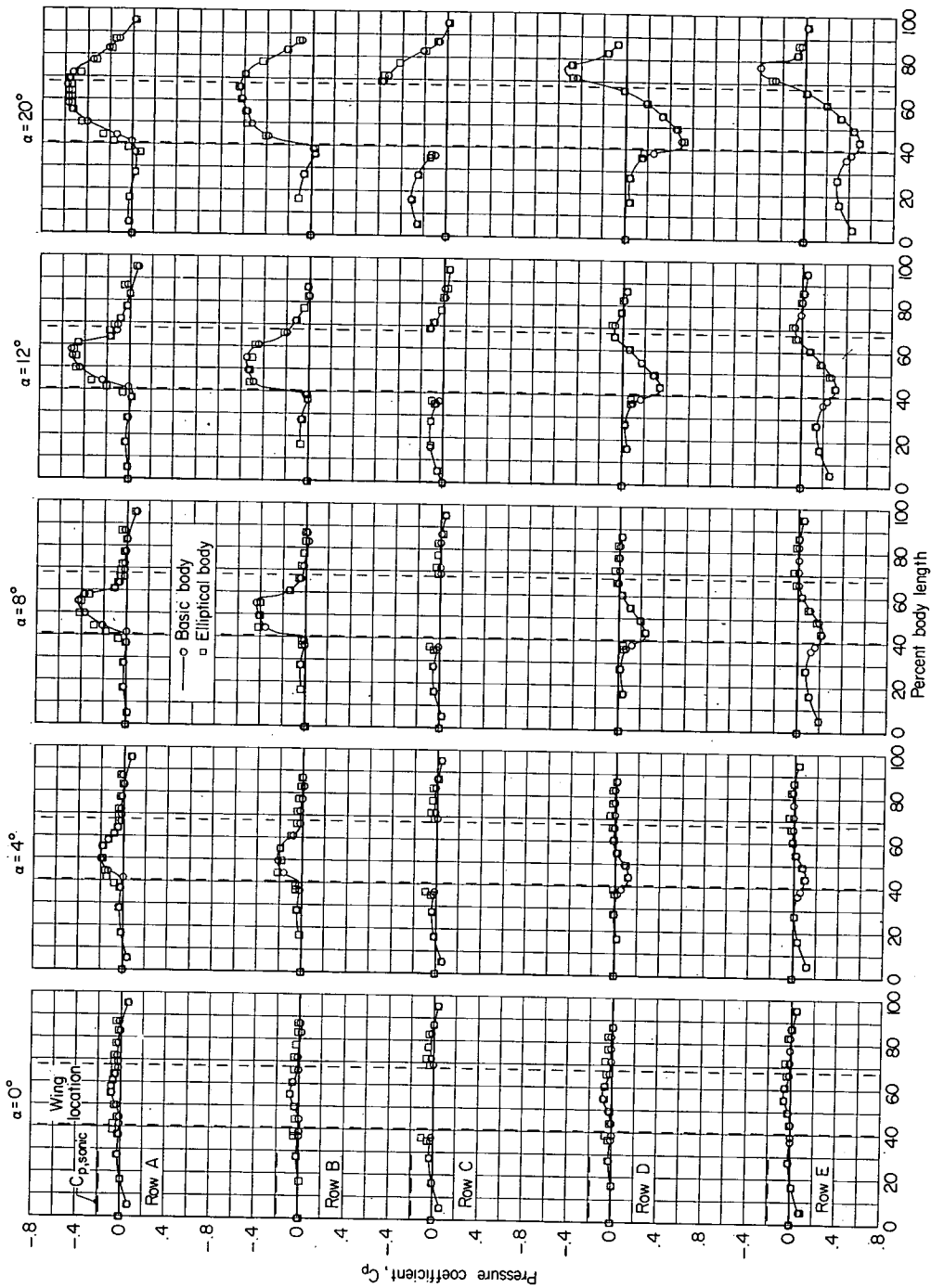
(g)  $M = 1.115$ .

Figure 4.- Concluded.



(a)  $M = 0.80$ .

Figure 5.- Pressure coefficients for bodies in presence of wing.



(b)  $M = 0.90$ .

Figure 5.- Continued.

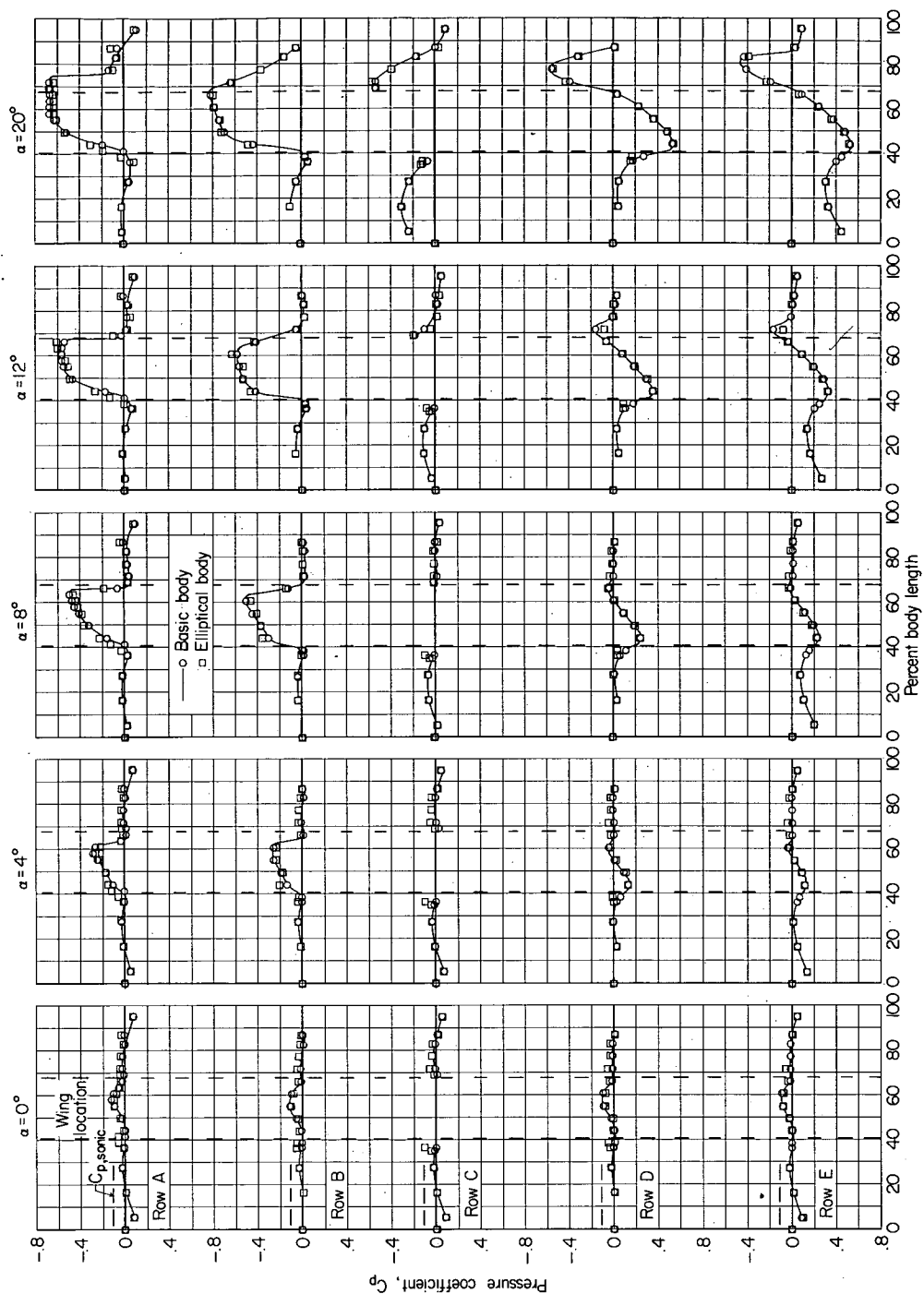
(c)  $M = 0.94$ .

Figure 5.- Continued.

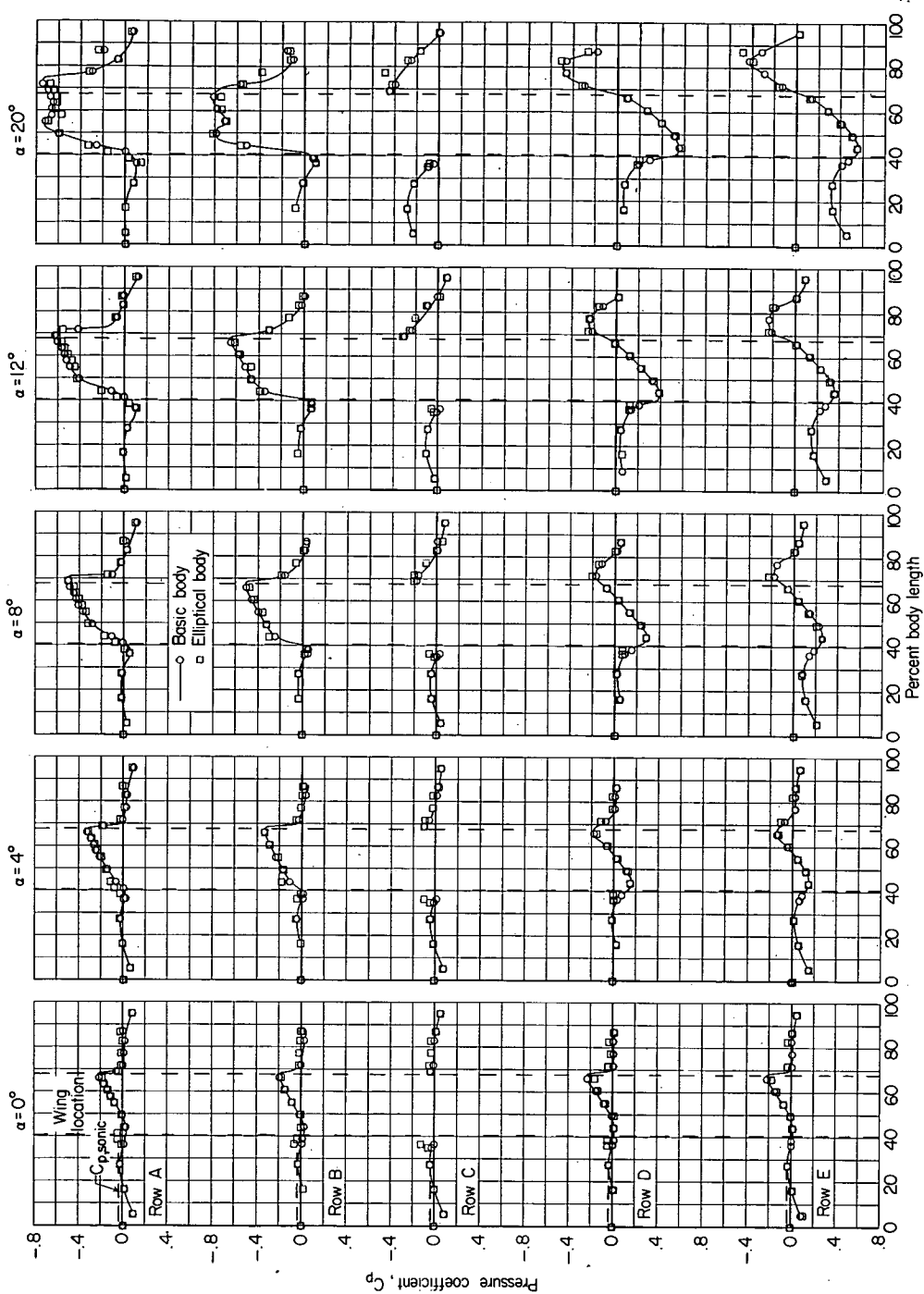
(d)  $M = 0.98$ .

Figure 5.- Continued.



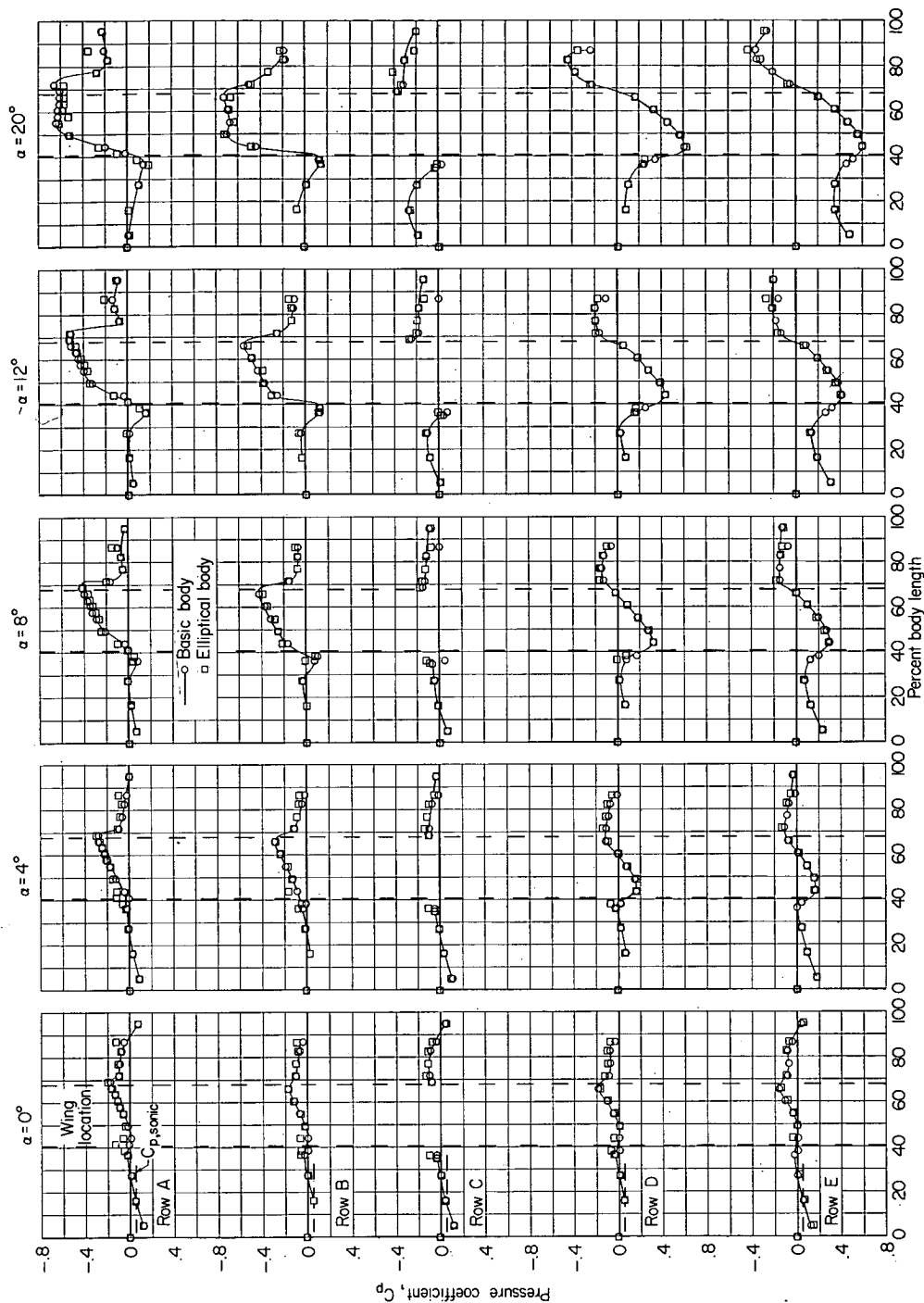
(e)  $M = 1.03$ .

Figure 5.- Continued.

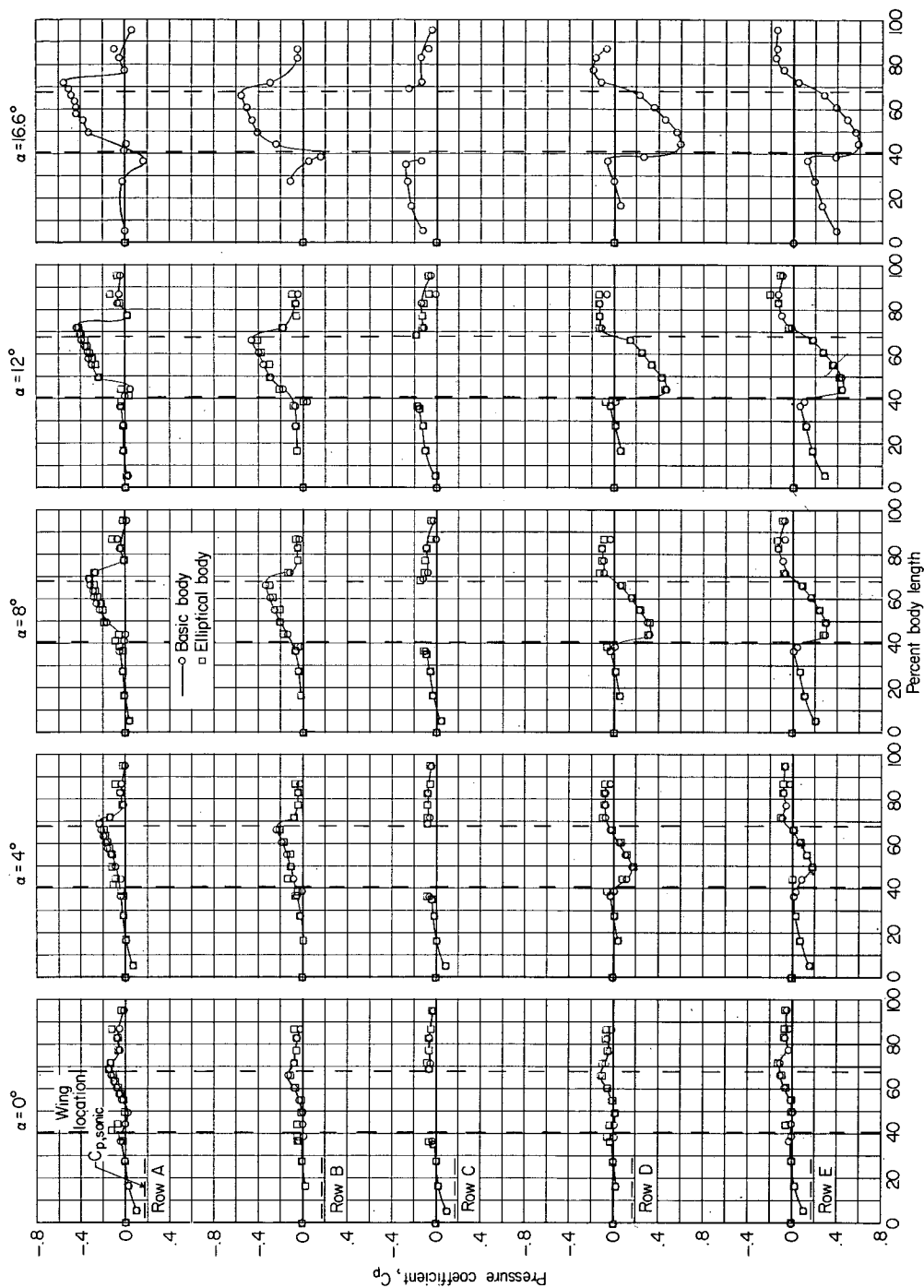
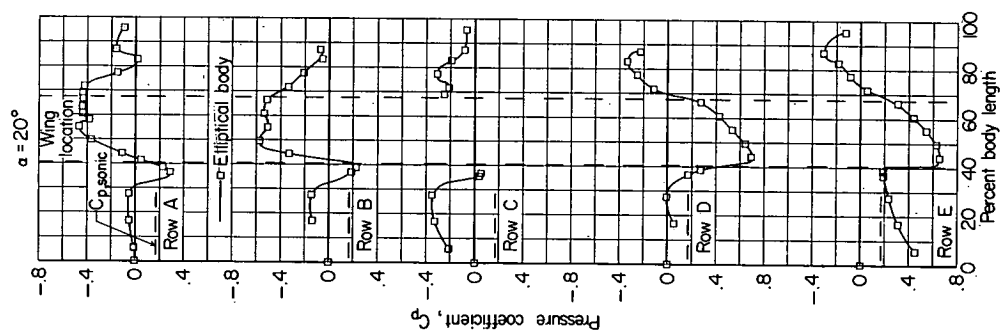
(f)  $M = 1.115$ .

Figure 5.- Continued.



(g)  $M = 1.115$ .

Figure 5.- Concluded.

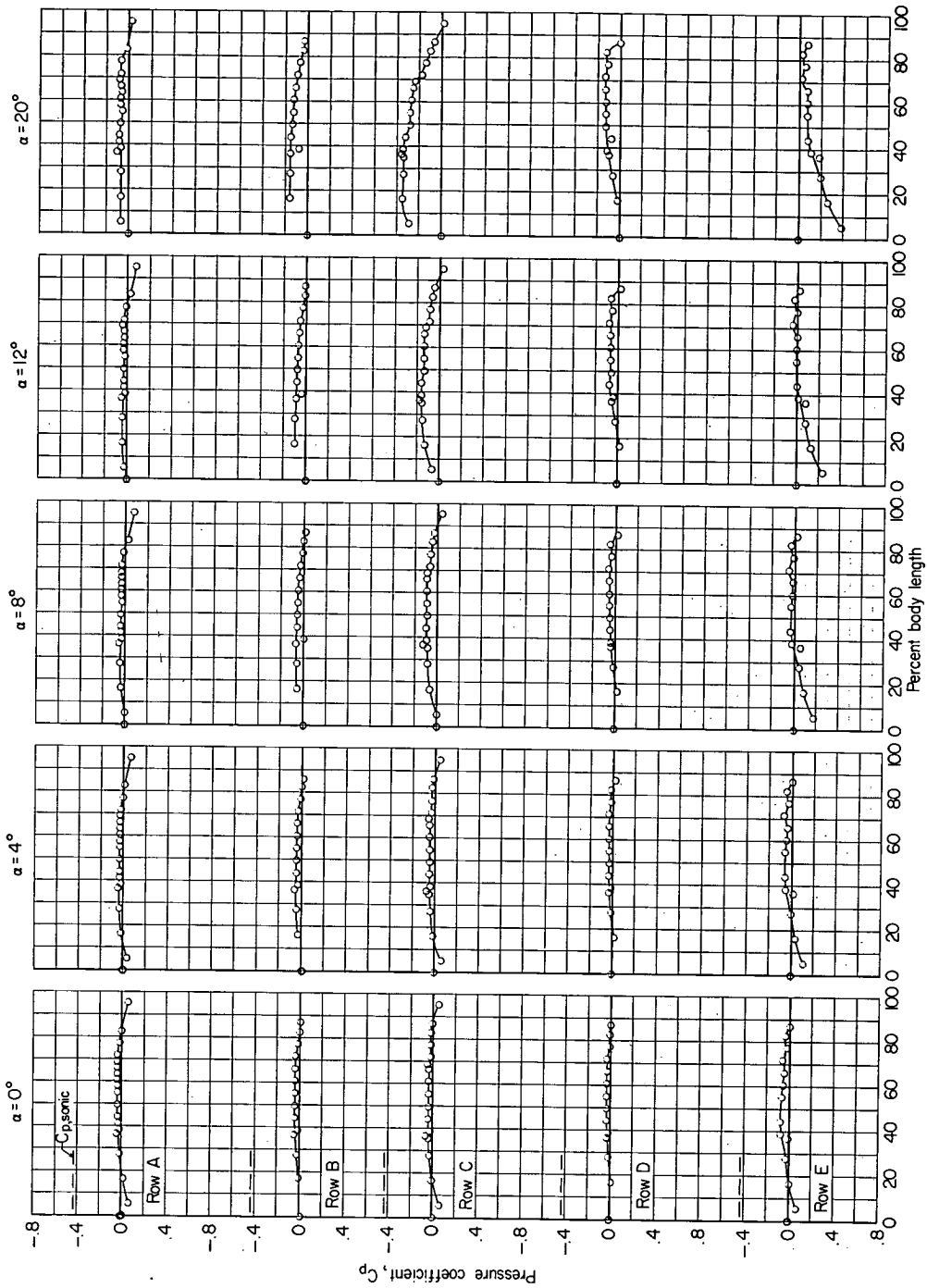
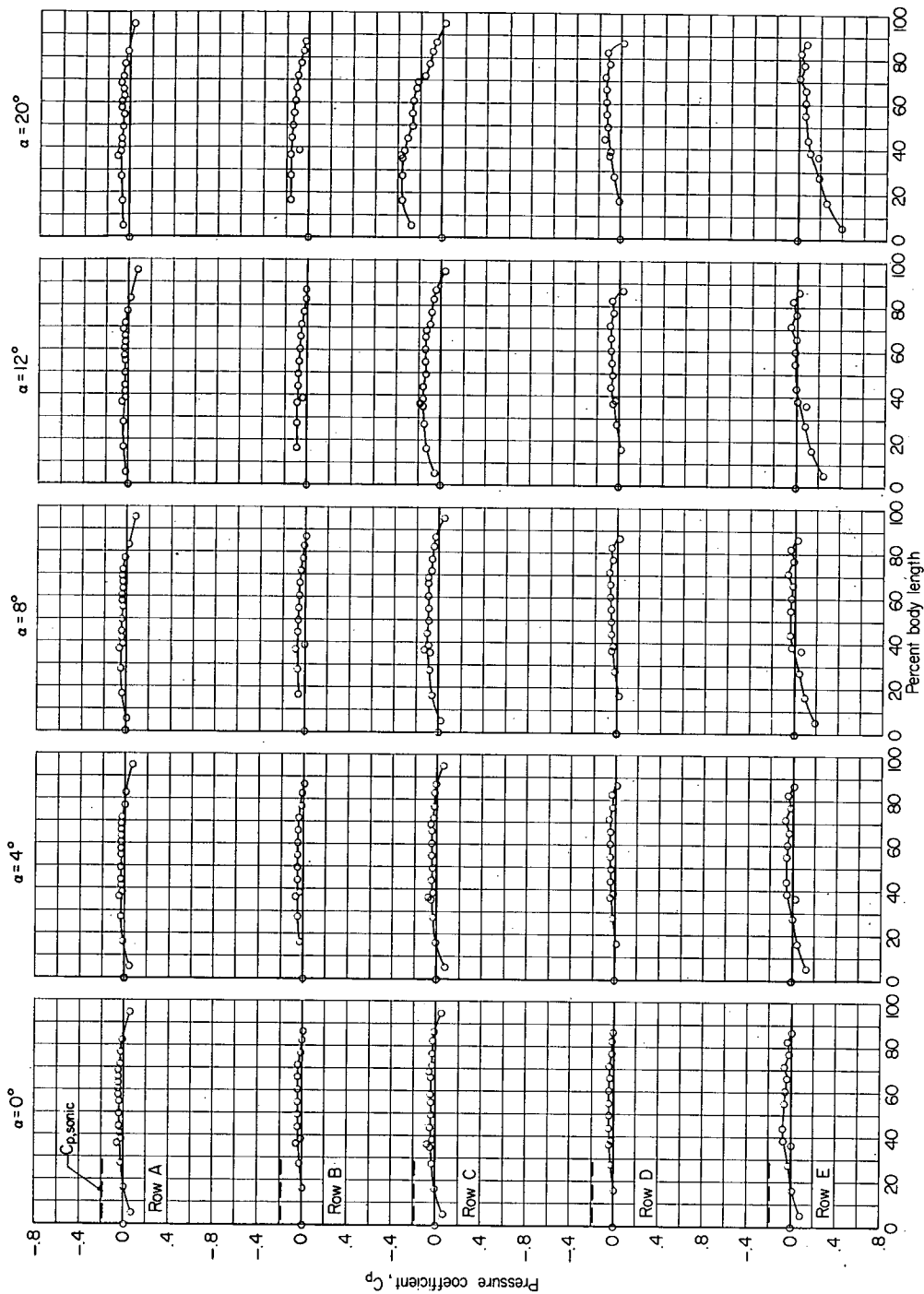
(a)  $M = 0.80$ .

Figure 6.- Pressure coefficients for basic body alone.



(b)  $M = 0.90$ .

Figure 6.- Continued.

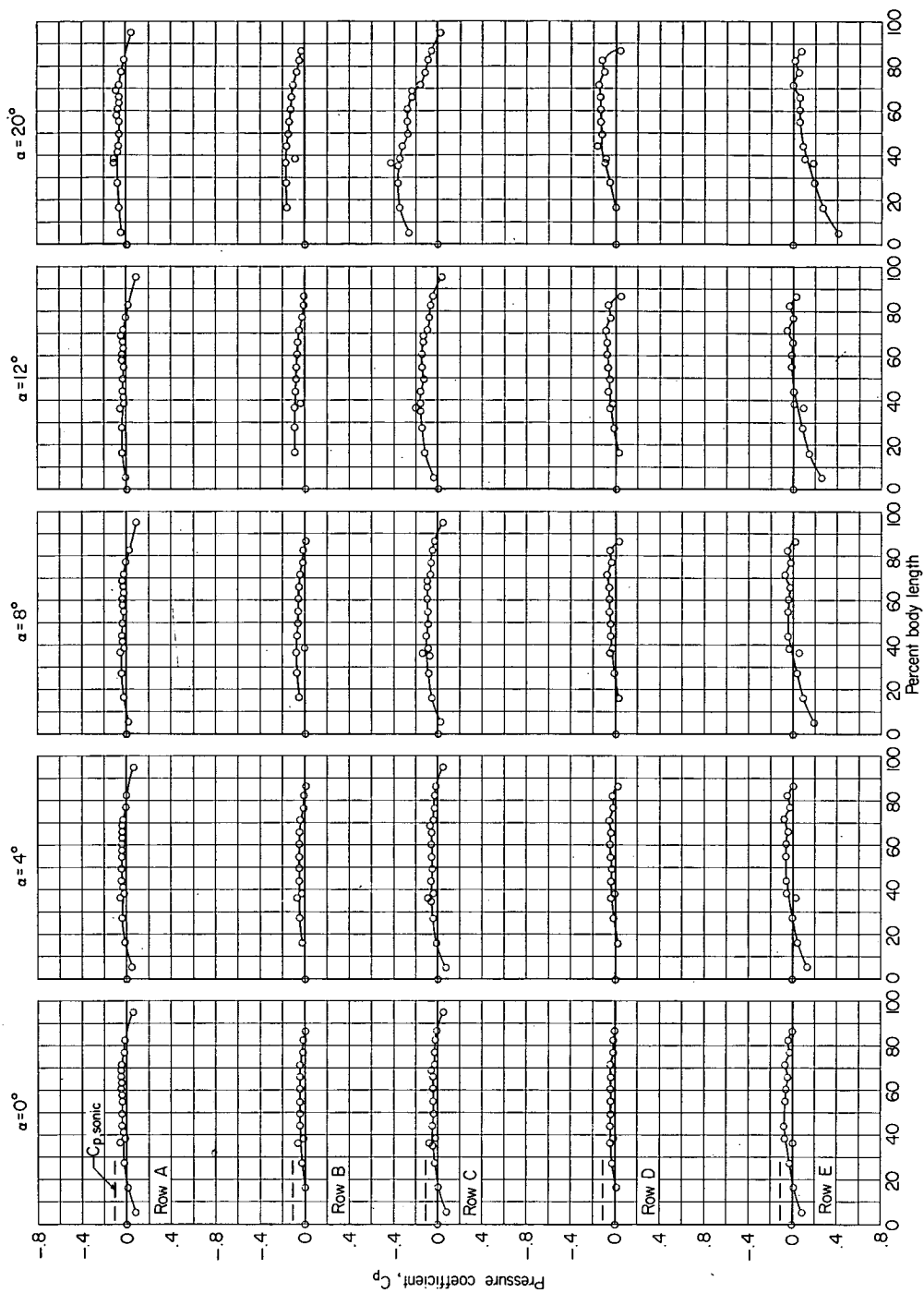
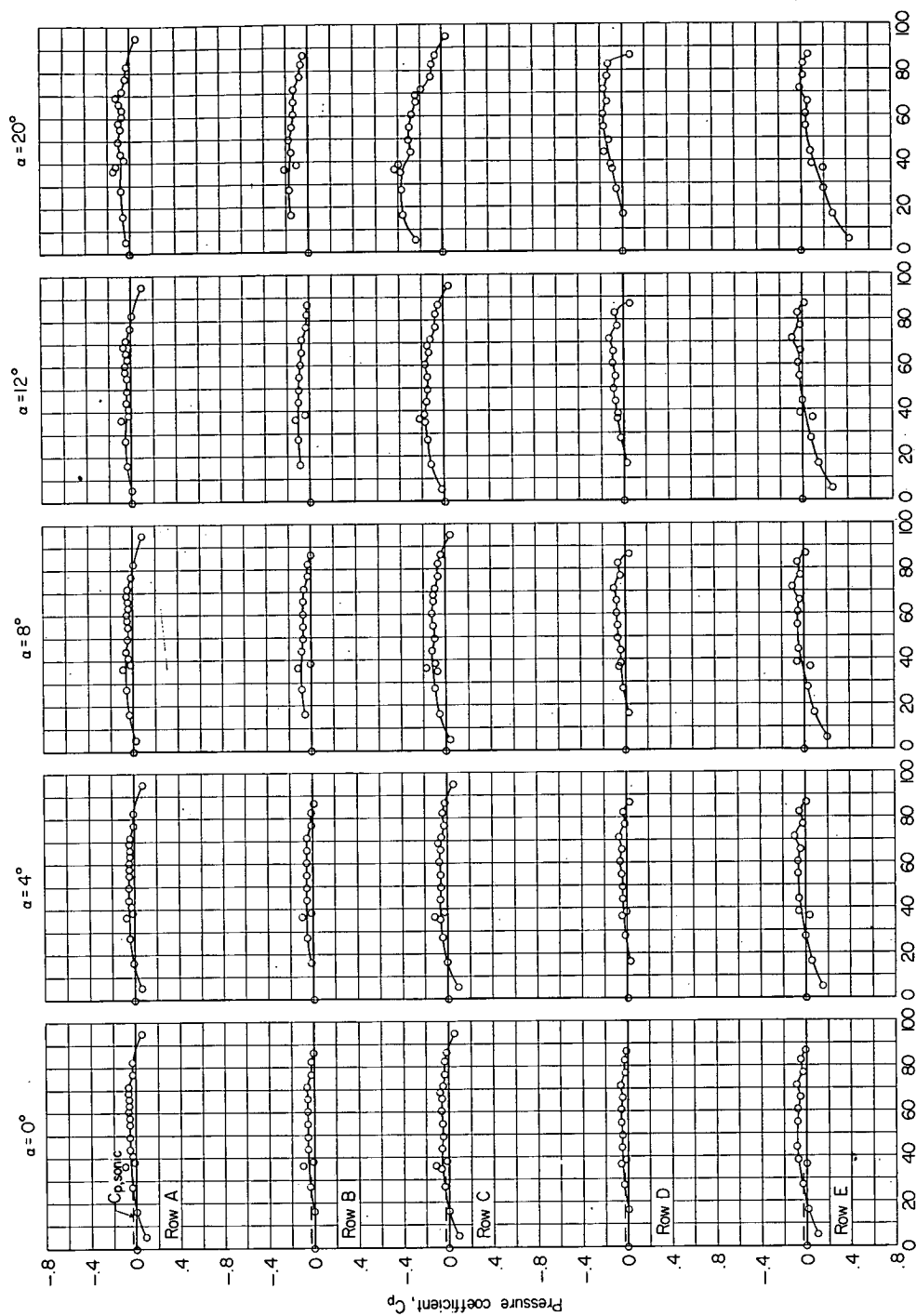
(c)  $M = 0.94$ .

Figure 6.- Continued.



(a)  $M = 0.98$ .

Figure 6.- Continued.

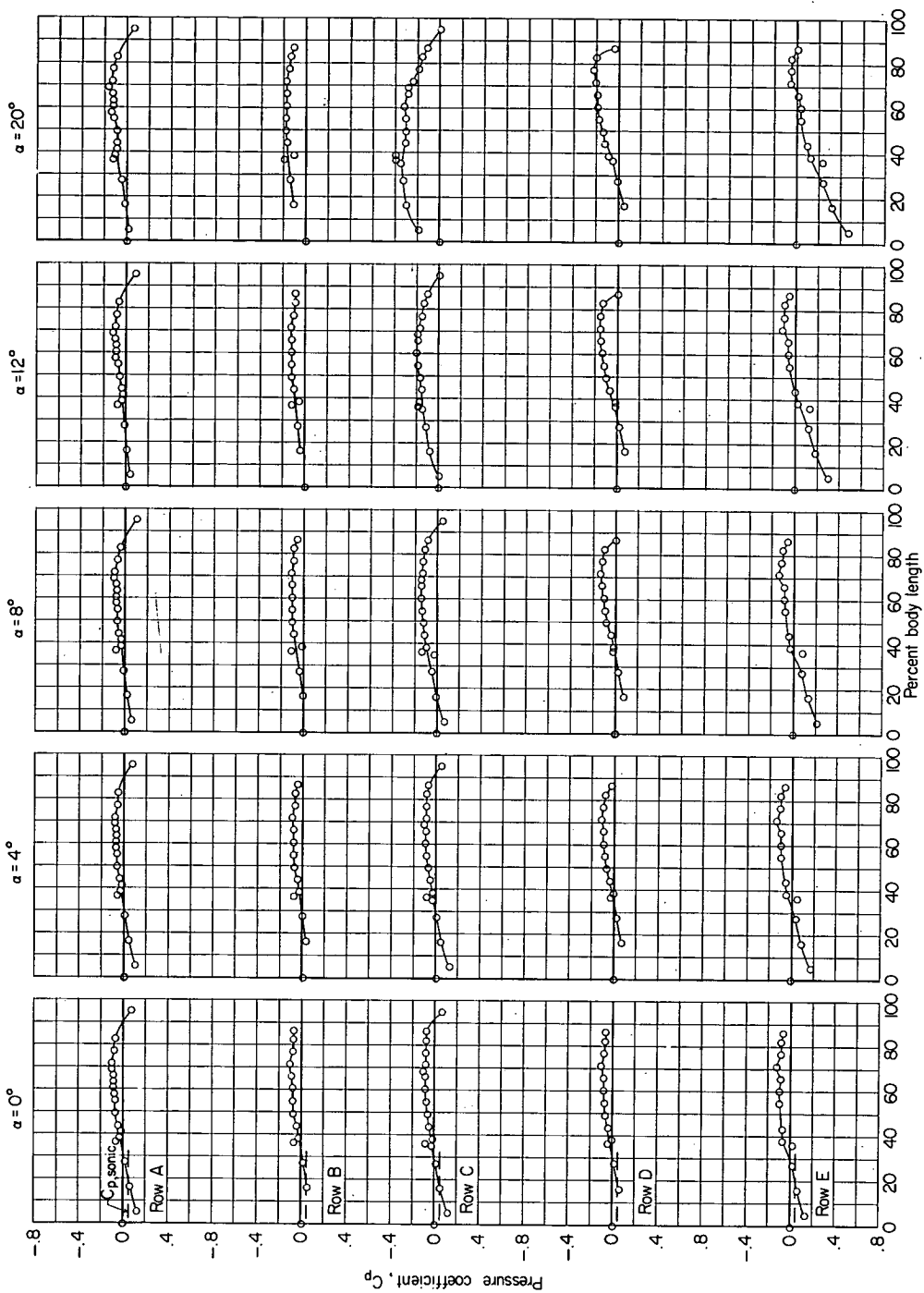
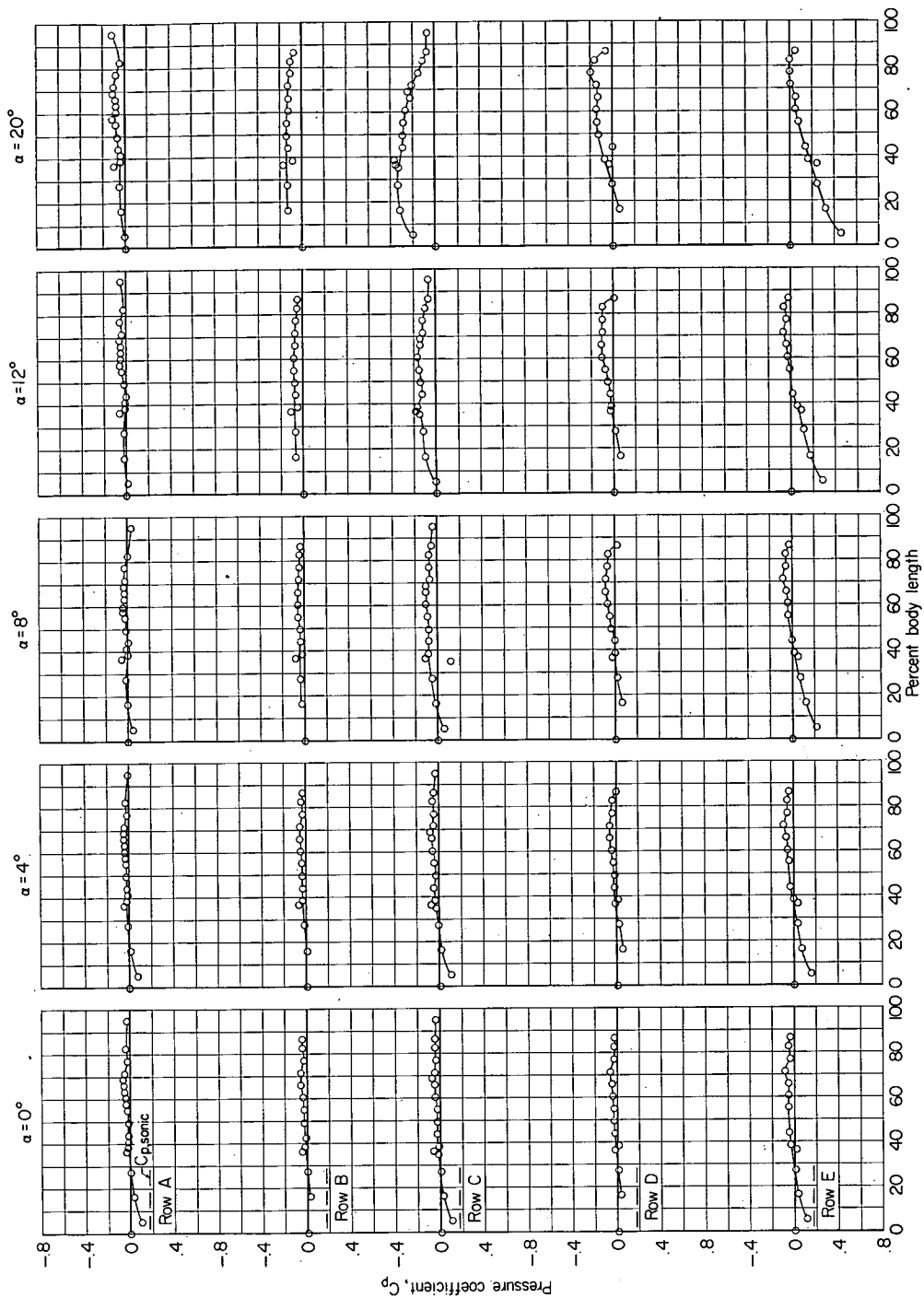
(e)  $M = 1.03$ .

Figure 6.- Continued.





(f)  $M = 1.115$ .

Figure 6.- Concluded.

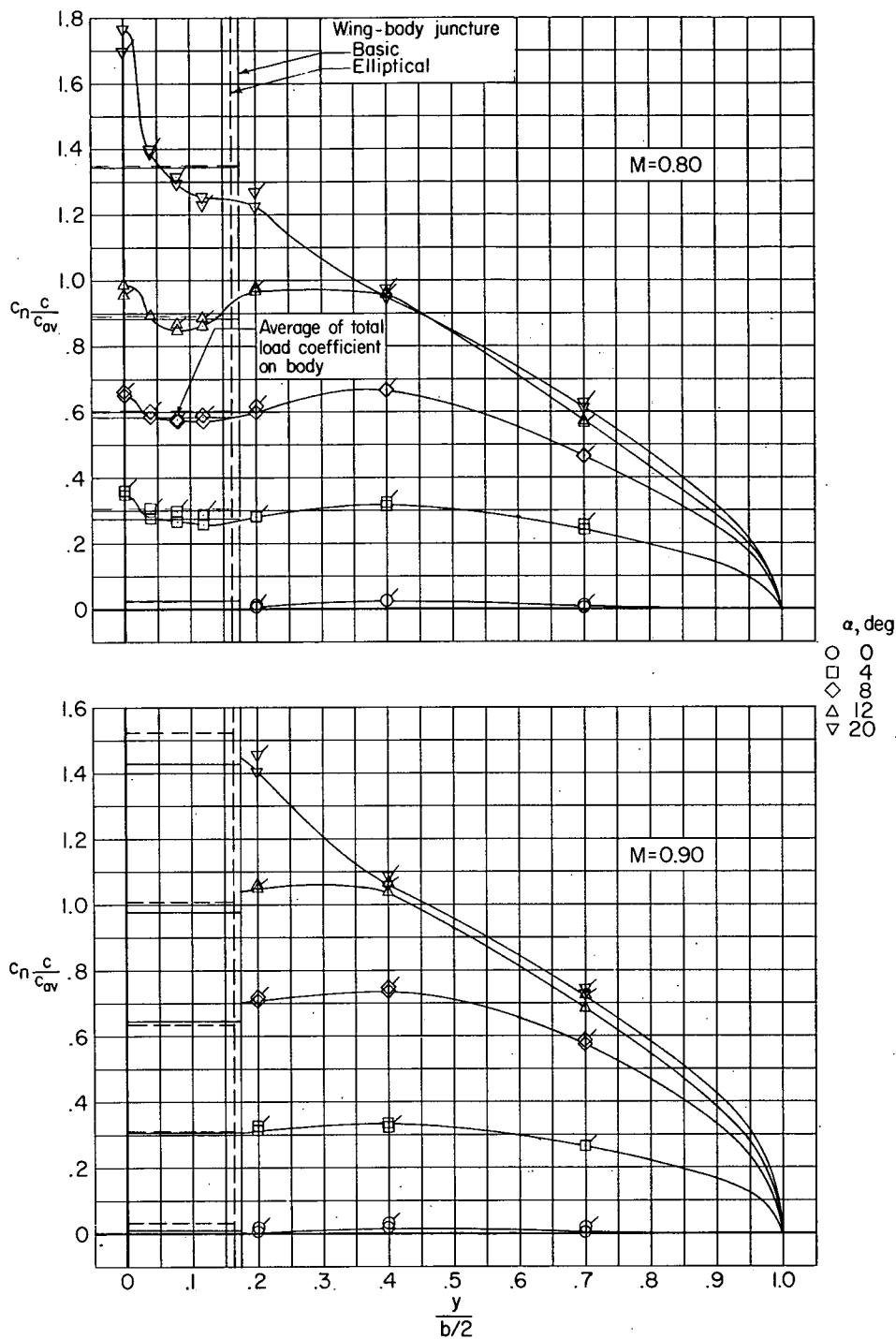


Figure 7.- Spanwise load distributions for two configurations. (Elliptical-body configuration is indicated by flagged symbols.)

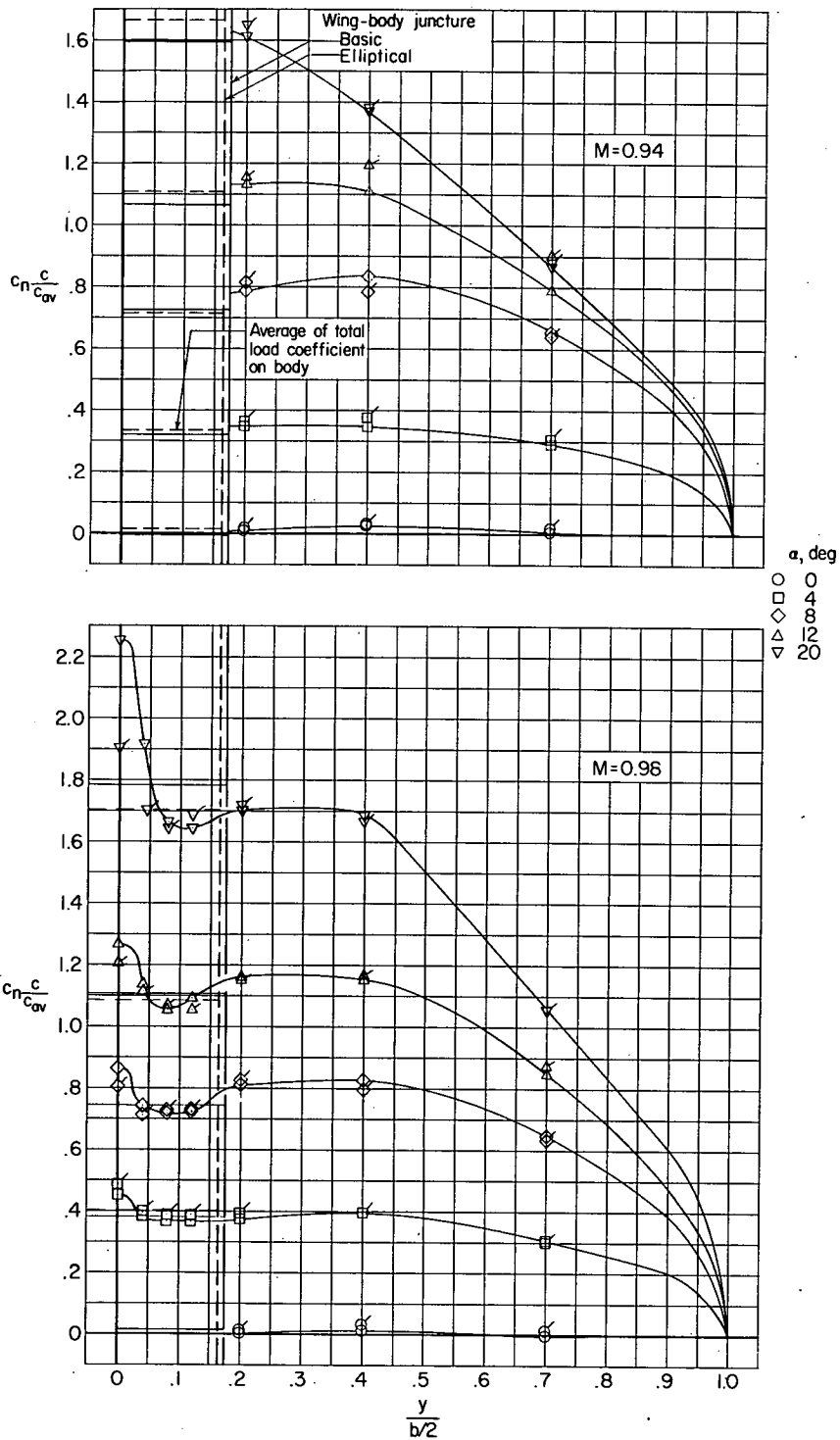


Figure 7.- Continued.

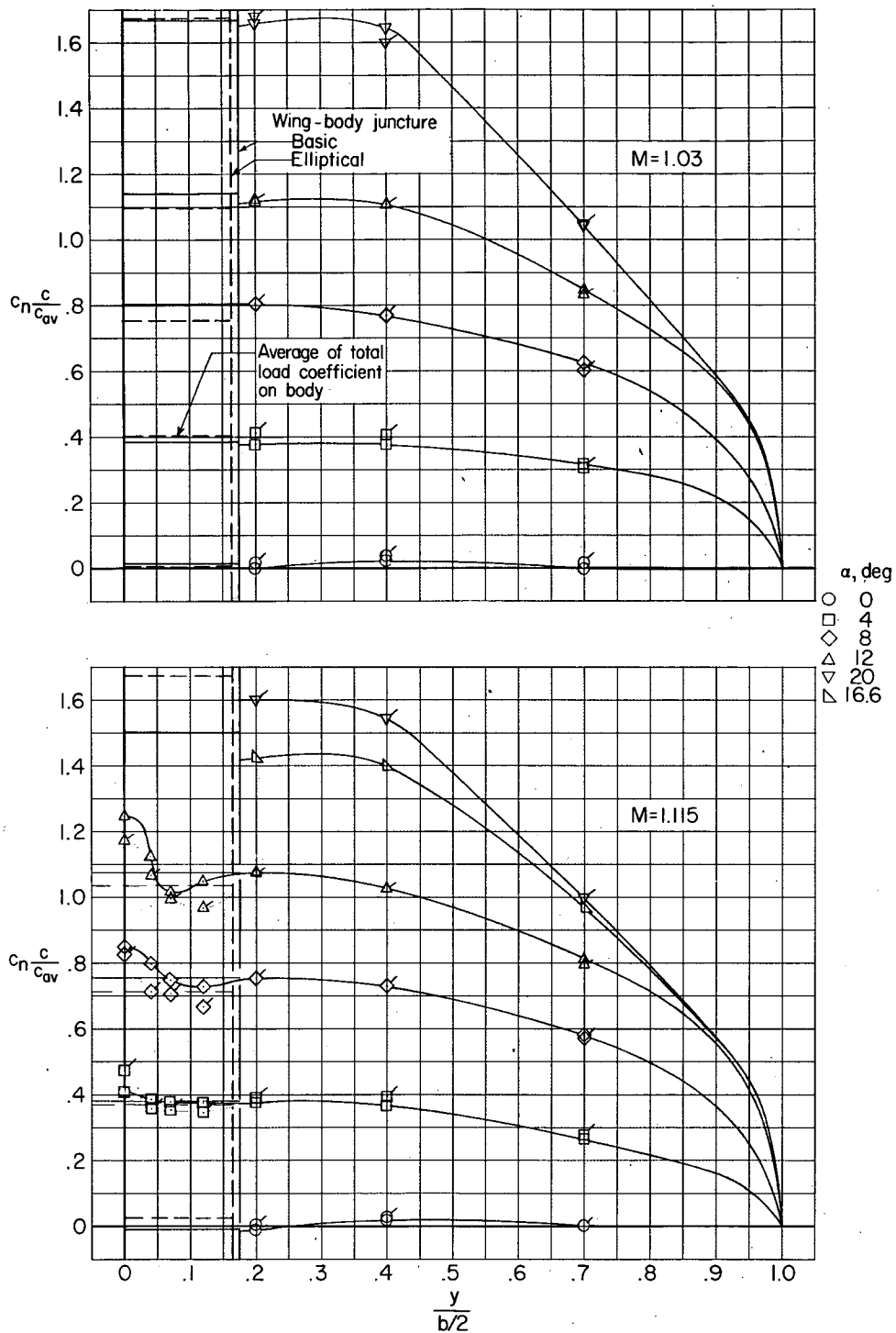


Figure 7.- Concluded.

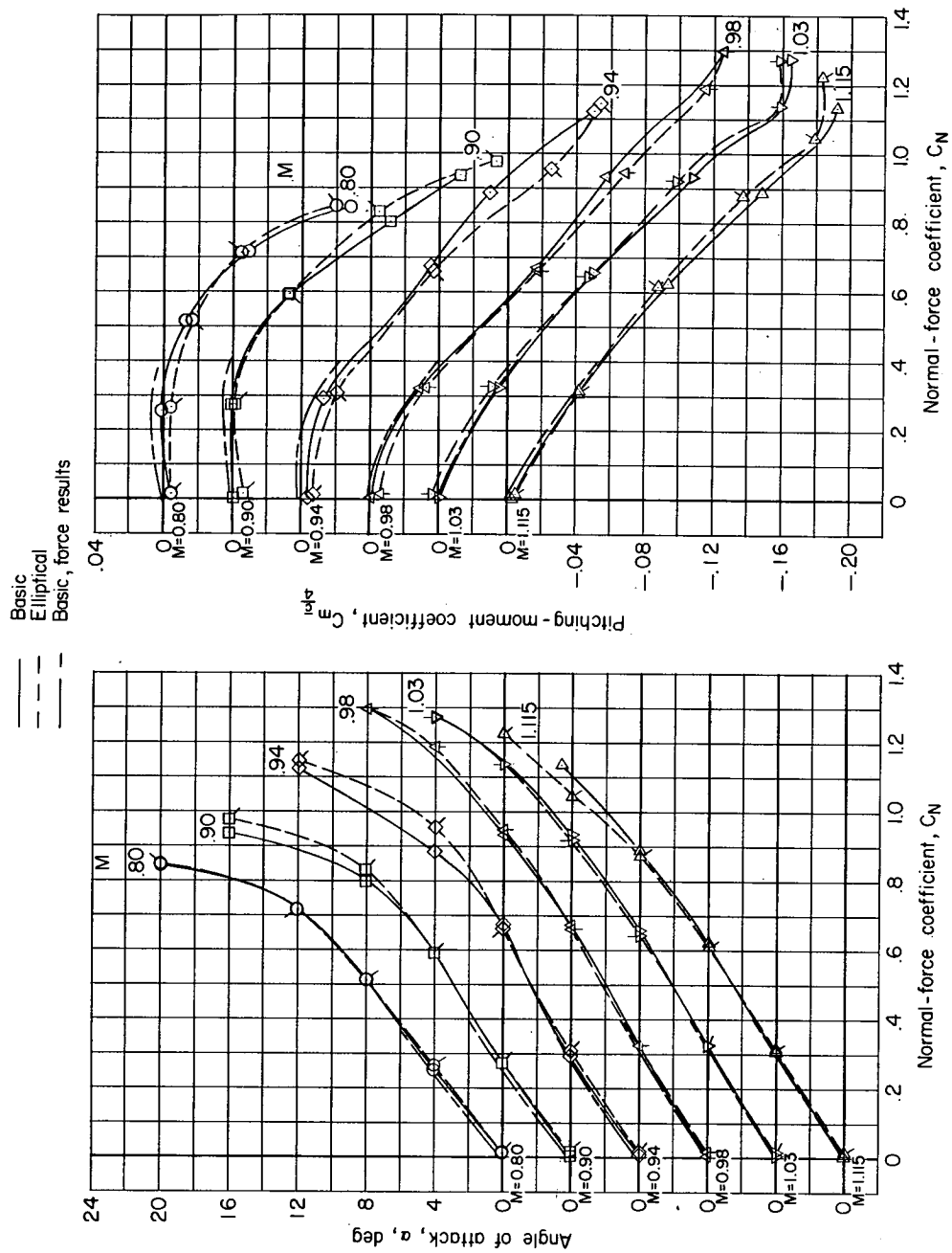


Figure 8.- Normal-force and pitching-moment characteristics of wing-body configurations. (Elliptical-body configuration is indicated by flagged symbols.)

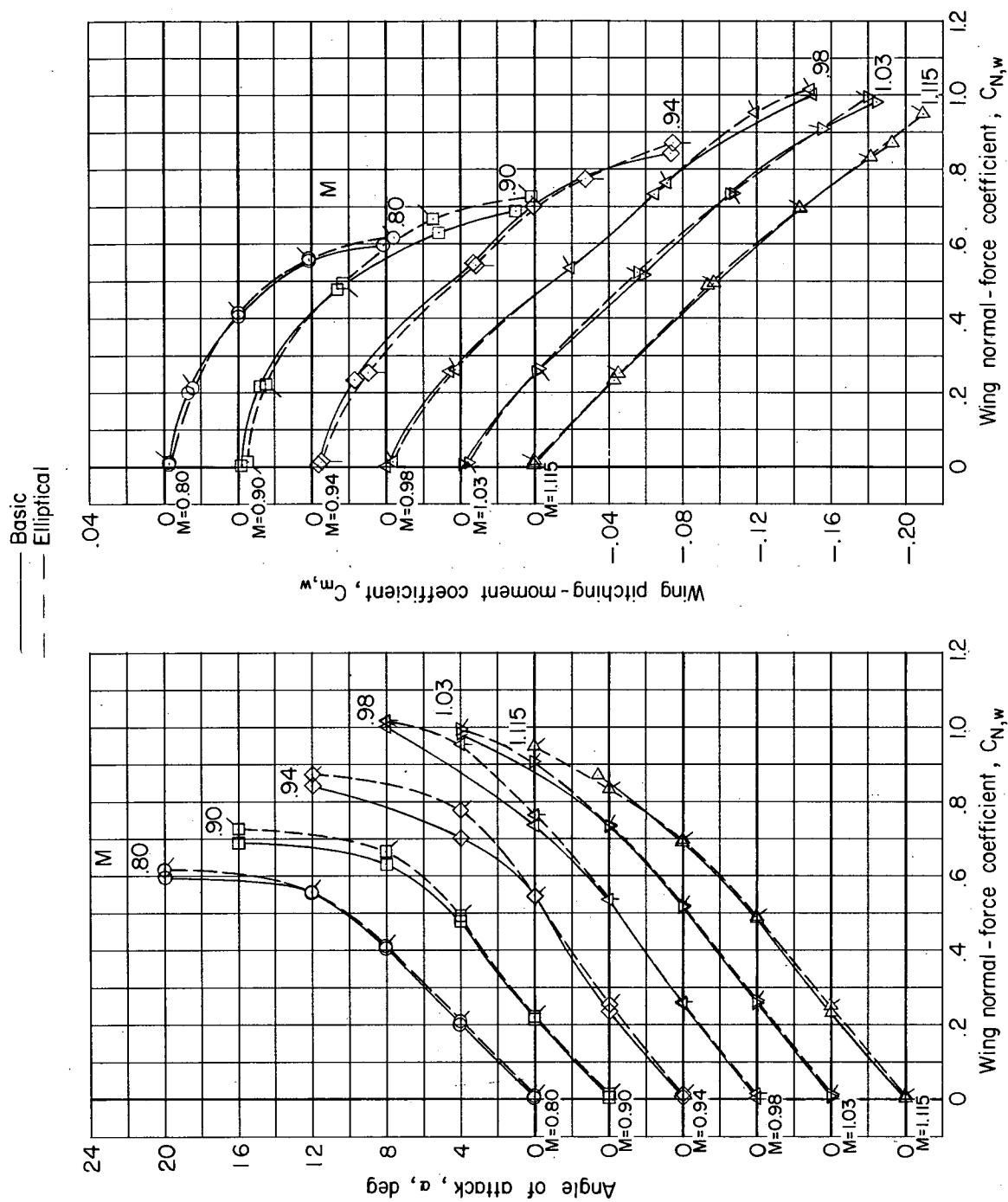


Figure 9.- Normal-force and pitching-moment characteristics of wing in the presence of body. (Elliptical-body configuration is indicated by flagged symbols.)

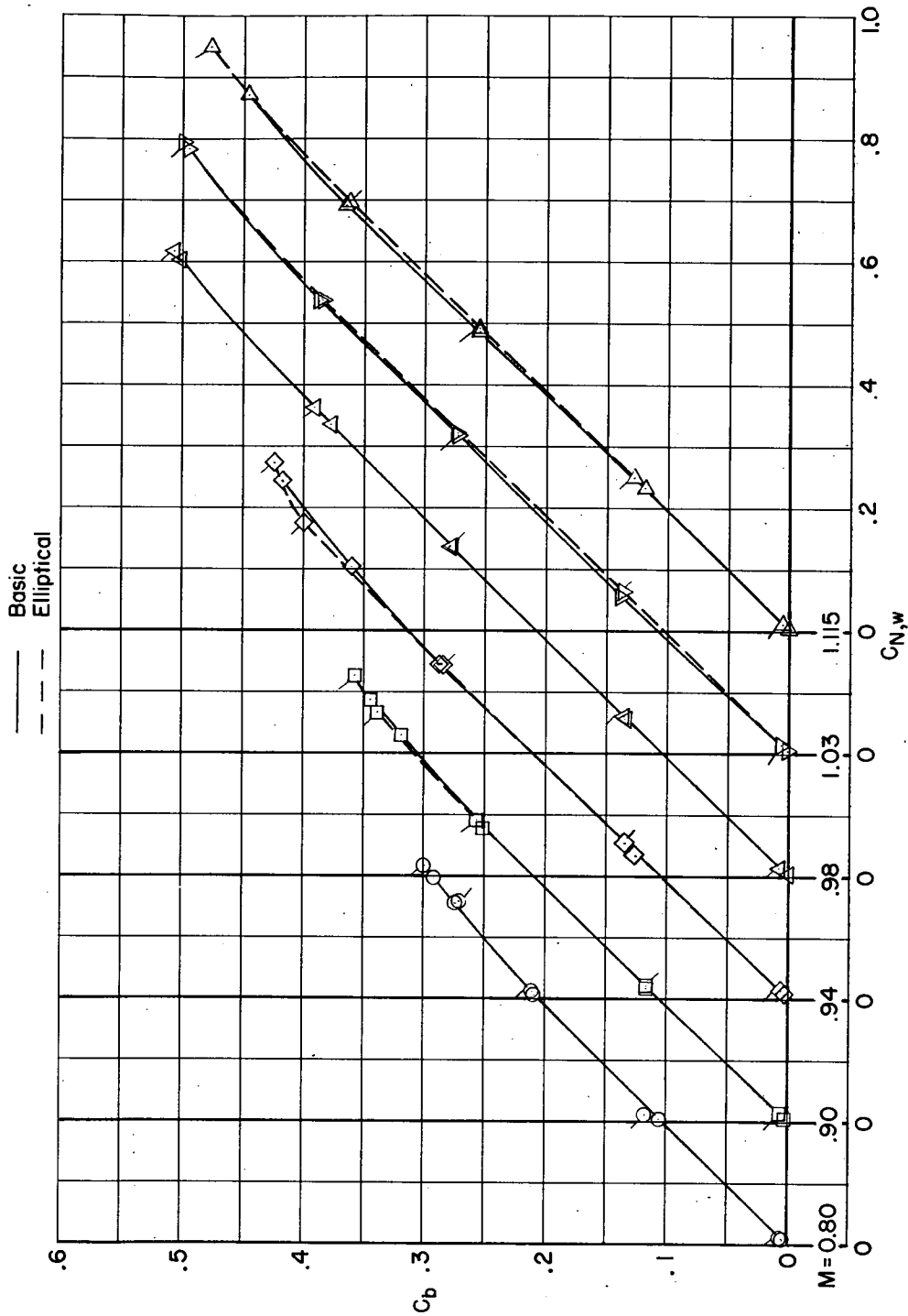
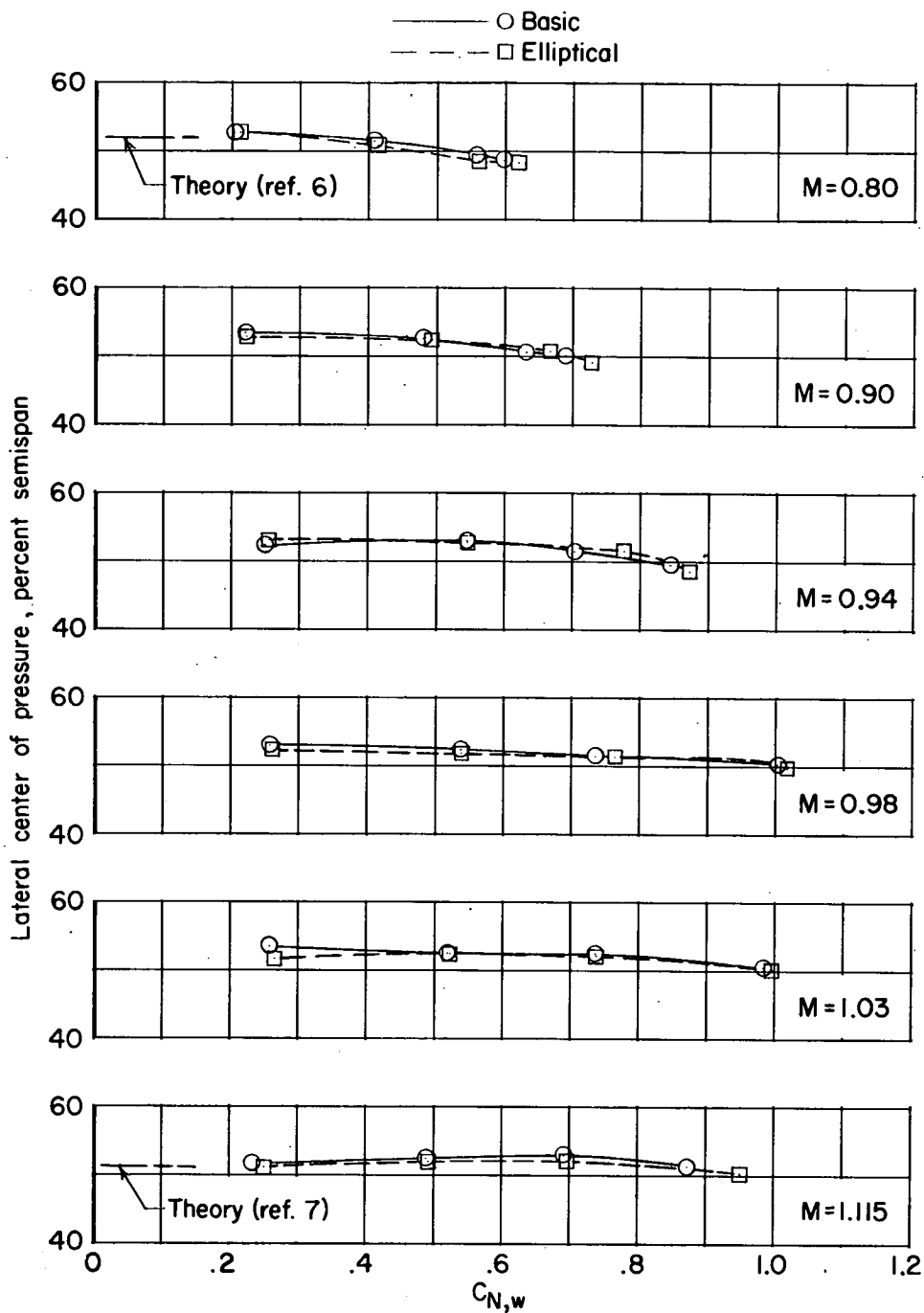


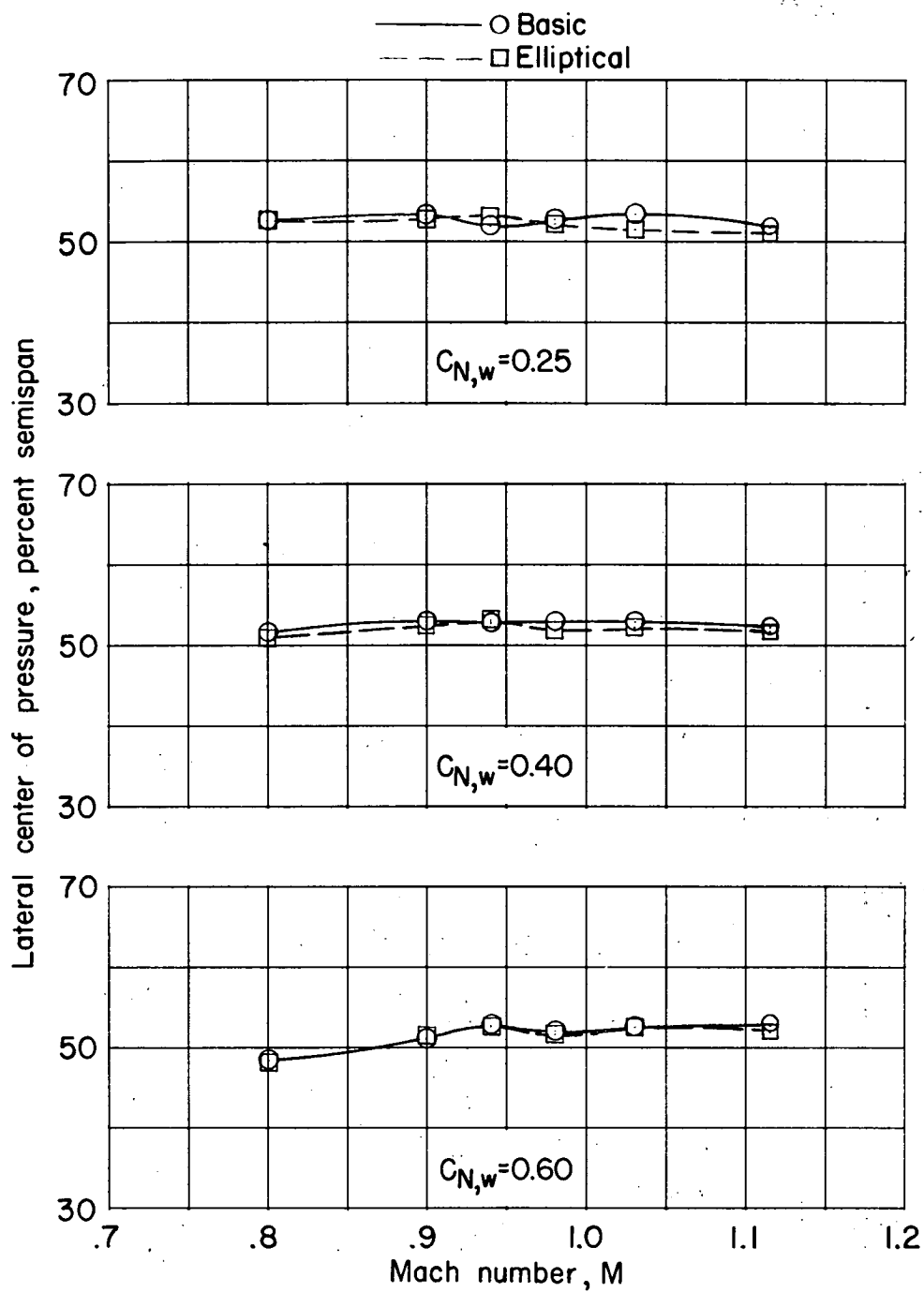
Figure 10.- Variation of wing bending-moment coefficient (referred to body center line) with normal-force coefficient for several Mach numbers. (Elliptical-body configuration is indicated by flagged symbols.)



(a) Variation with wing normal-force coefficient.

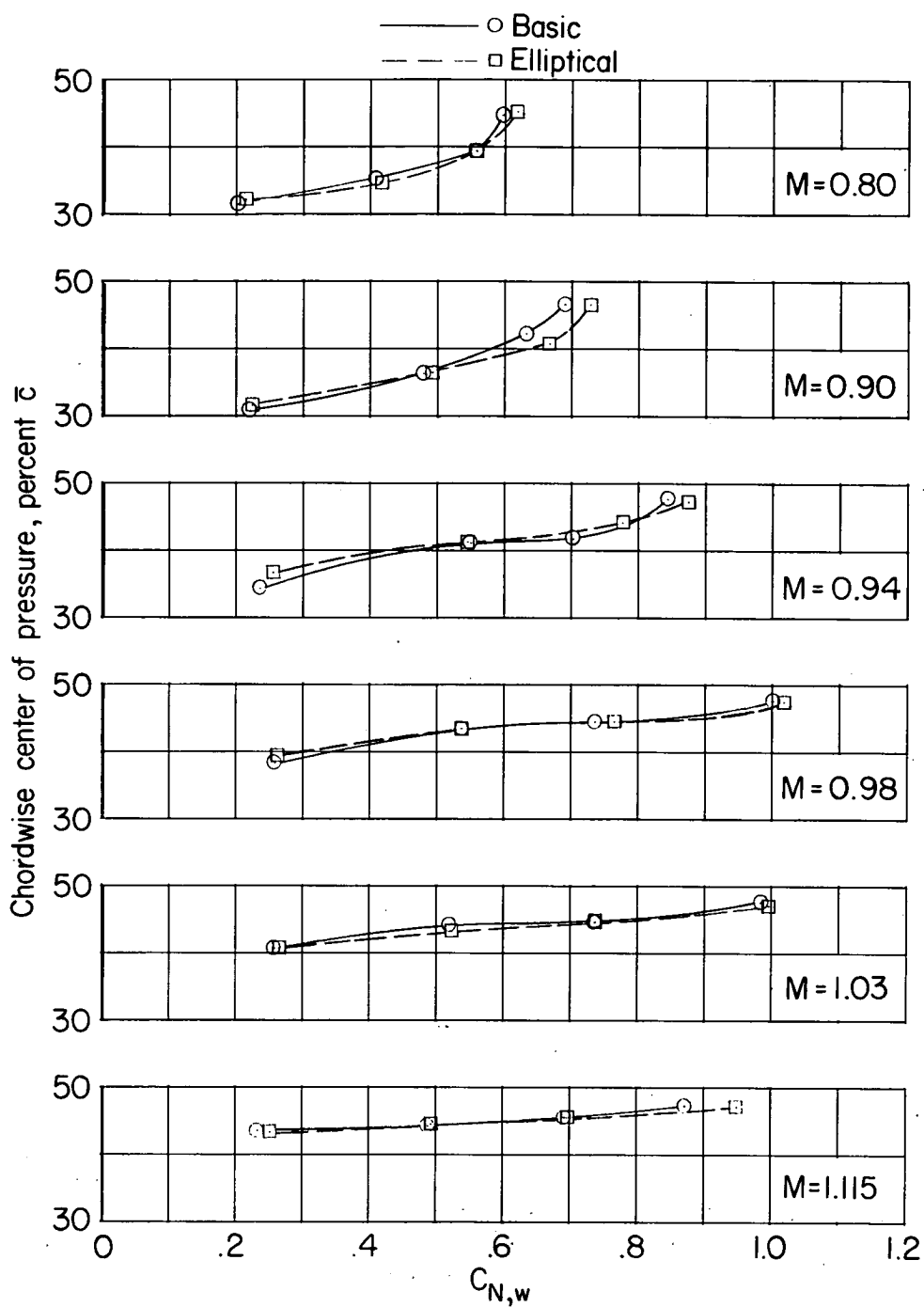
Figure 11.- Lateral center-of-pressure position.





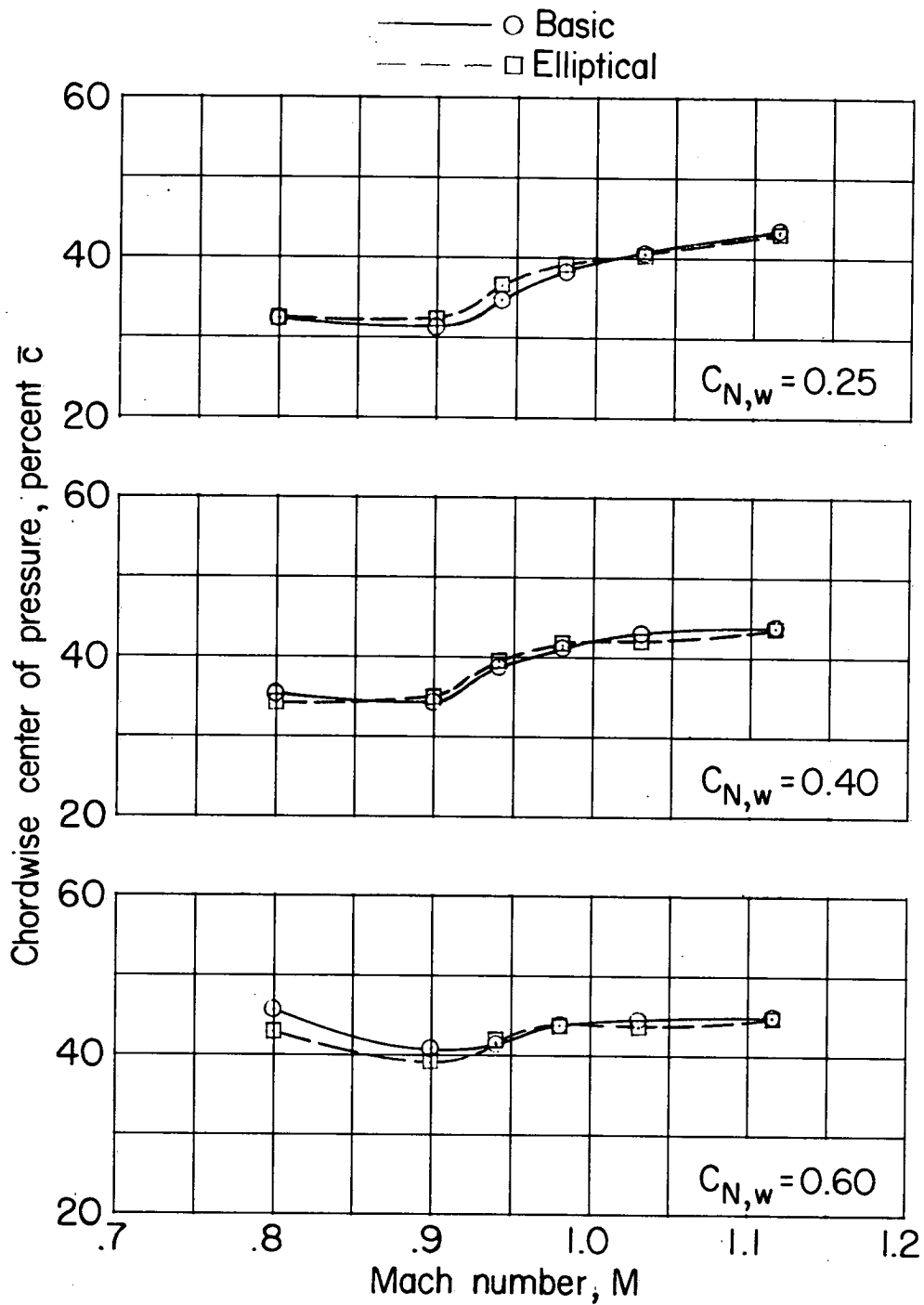
(b) Variation with Mach number.

Figure 11.- Concluded.



(a) Variation with wing normal-force coefficient.

Figure 12.- Chordwise center-of-pressure position.



(b) Variation with Mach number.

Figure 12.- Concluded.

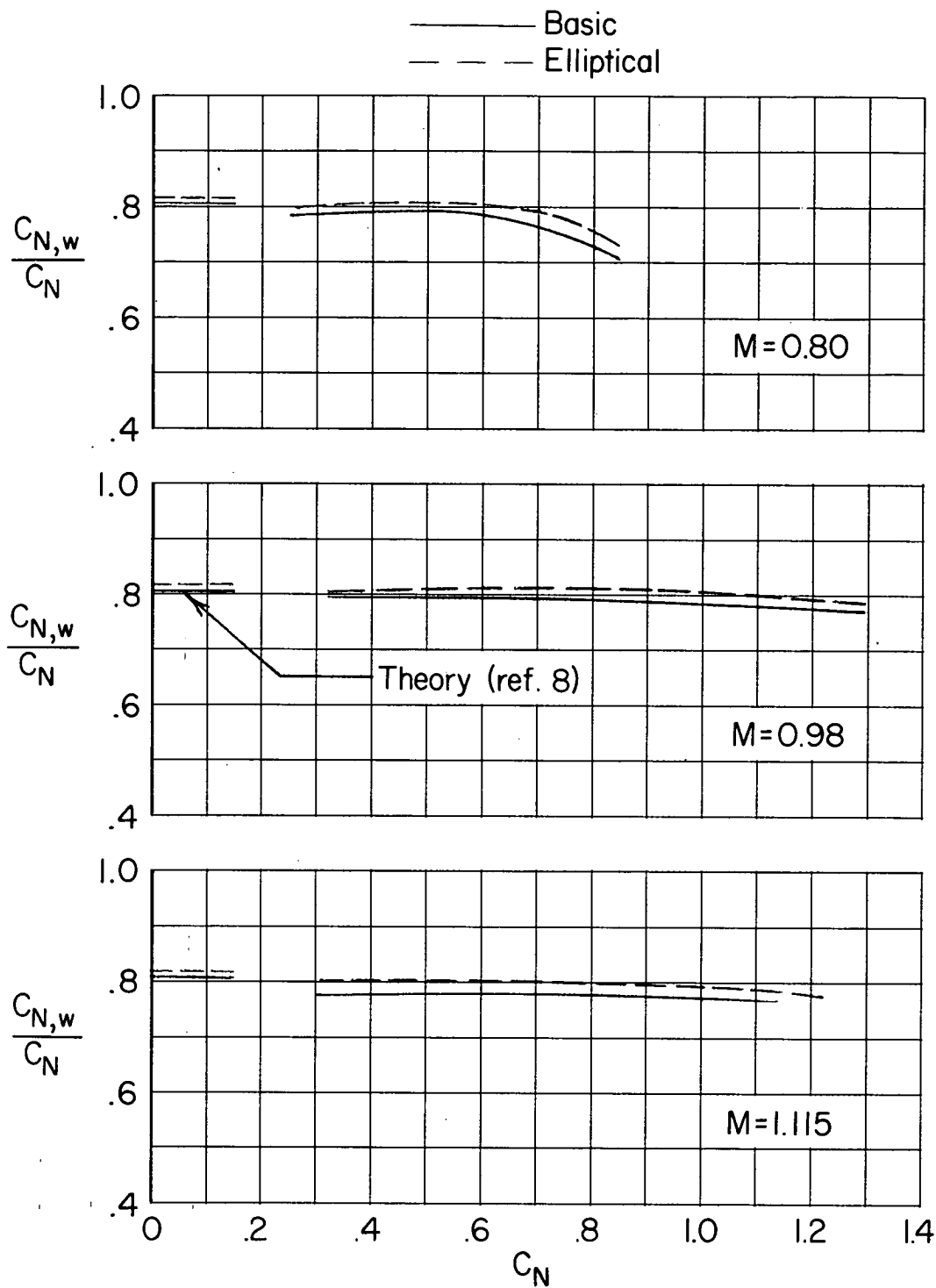


Figure 13.- Part of total load carried by wing.

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